THE ATS-F and -G DATA BOOK

ATS 6
7

12-71 F. Schmidlin

GENERAL INFORMATION

SPACECRAFT

LAUNCH VEHICLE

ATS OPERATIONS

TRAJECTORY and ORBIT

GROUND TERMINALS

TECHNOLOGICAL EXPERIMENTS

SCIENTIFIC EXPERIMENTS

APPENDIX A
USEFUL INFORMATION

APPENDIX B
ACRONYMS

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CONTENTS

Section		Page
I	GENERAL INFORMATION	
	INTRODUCTION	I-1
	HISTORICAL BACKGROUND	I-3
	ATS-1 THROUGH -5 OPERATIONAL HISTORIES	I-5
	ATS-1 (ATS-B Before Launch)	I-5
	ATS-2 (ATS-A Before Launch)	I-5
	ATS-3 (ATS-C Before Launch)	I-6
	ATS-4 (ATS-D Before Launch)	I-6
	ATS-5 (ATS-E Before Launch)	I-7
	MISSION PROFILE	I-8
II	SPACECRAFT	
	STRUCTURE	II-1
	THERMAL CONTROL SUBSYSTEM	II-13
	Subsystem Design	II-13
	ATTITUDE CONTROL SUBSYSTEM	II-15
	Subsystem Description	II-21
	SPACECRAFT PROPULSION SUBSYSTEM	II-39
	ELECTRICAL POWER SUBSYSTEM	II-45
	Functional Operation	II-43
	Power Subsystem Capability Requirements	II-43
	Power Subsystem Protective Features	II-44
	COMMUNICATIONS SUBSYSTEM	II-54
	Antennas	II-55
	Physical Description	II-55
	Transponder	II-60

Section		Page
II	SPACECRAFT (Cont)	
	TELEMETRY AND COMMAND SUBSYSTEM	II-65
	Command	II-65
	Telemetry	II-74
	REFLECTOR	II-77
	HARNESS	II-77
	RF INTERFEROMETER SUBSYSTEM	II-83
	System Description	II-83
III	LAUNCH VEHICLE	
	TITAN III-C DESCRIPTION	III-1
	Stage 0	III-1
	Stage I	III-3
	Stage II	III-4
	Transtage	III-4
	UHF Systems	III-5
	Hydraulics System	III-5
	Guidance, Control, and Instrumentation Systems	III-5
	Ordnance	III-6
	Fairing	III-6
	Launch	III-7
	ATS-F AND -G MISSION SPECIFICATION	III-8
IV	ATS OPERATIONS	
	INTRODUCTION	IV-1
	ATS OPERATIONS CONTROL CENTER	IV-2

CONTENTS (Cont)

Section		Page
IV	ATS OPERATIONS (Cont)	
	NASA COMMUNICATIONS NETWORK	IV-4
	COMPUTATION DIVISION	IV-5
	INFORMATION PROCESSING DIVISION	IV-5
V	TRAJECTORY AND ORBIT	
	DESCRIPTION OF MANEUVER	V-1
	Launch Sequence	V-1
	Aquisition Sequence	V-3
	Operational Modes	V-4
	DESCRIPTION OF CONTROL SYSTEM	V-6
	Introduction	V-6
	Sensors	V-9
	Controllers	V-11
	Torquers	V-12
VI	GROUND TERMINALS	
	INTRODUCTION	VI-1
	Rosman Ground Station	VI-2
	Mojave Ground Station	VI-12
	Transportable Ground Station	VI-14
	Mobile Terminal	VI-17
VII	TECHNOLOGICAL EXPERIMENTS	
	VERY HIGH RESOLUTION RADIOMETER EXPERIMENT	VII-1
	POSITION LOCATION AND AIRCRAFT COMMUNICATION EXPERIMENT	VII-2

CONTENTS (Cont)

Section		Page
VII	TECHNOLOGICAL EXPERIMENTS (Cont)	
	INSTRUCTIONAL TELEVISION EXPERIMENT	VII-4
	TELEVISION RELAY USING SMALL TERMINALS	VII-11
	RADIO FREQUENCY INTERFERENCE EXPERIMENT	VII-14
	MILLIMETER WAVE EXPERIMENT	VII-15
	TRACKING AND DATA RELAY EXPERIMENT	VII-19
	SPACECRAFT ATTITUDE PRECISION POINTING & SLEWING ADAPTIVE CONTROL EXPERIMENT . .	VII-21
	CESIUM BOMBARDMENT ION ENGINE EXPERIMENT	VII-21
	ATS-F 13- AND 18-GHz PROPAGATION EXPERIMENT	VII-21
	Introduction	VII-21
	Objectives of the Experiment	VII-22
	Description of the System	VII-23
	Operation and Propagation Data	VII-24
	LASER RETROREFLECTOR EXPERIMENT	VII-29
	Introduction	VII-29
	Objectives of the Experiment	VII-29
	EDUCATIONAL TELEVISION EXPERIMENT	VII-34
	Introduction	VII-34
	System Capability	VII-35
	ATS-F Implementation	VII-35

CONTENTS (Cont)

Section		Page
VIII	SCIENTIFIC EXPERIMENTS	
	RADIO BEACON EXPERIMENT	VIII-1
	ENVIRONMENTAL MEASUREMENTS EXPERIMENT . .	VIII-2
	Low-Energy Proton-Electron	VIII-2
	Low-Energy Protons	VIII-6
	Solar Cosmic Ray	VIII-6
	Auroral Particles	VIII-7
	Particle Acceleration Measurements	VIII-7
	Magnetometer	VIII-8
	Solar Cell Radiation Damage	VIII-8
	Omnidirectional Spectrometer	VIII-9
	APPENDIX A	
	Useful Information	A-1
	APPENDIX B	
	Acronyms	B-1

ILLUSTRATIONS

Figure		Page
I-1	ATS-F and -G Launch and Acquisition Sequence	I-11
II-1	ATS-F Configuration in Orbit	II-1
II-2	Spacecraft Structural Assemblies	II-2
II-3	Structural Hub	II-3
II-4	Hub and Bridge Truss	II-4
II-5	Spacecraft Deployed Configuration	II-5
II-6	Spacecraft Launch Configuration	II-7
II-7	Solar Panel Deployment	II-8
II-8	Earth Viewing Module Showing Thermal Louvers	II-10
II-9	Typical Heat Pipe Installation	II-12
II-10	Simplified ACS Block Diagram	II-13
II-11	ACS Block Diagram	II-21
II-12	Analog Backup Mode Block Diagram	II-23
II-13	Attitude Control System Component Location	II-29
II-14	Interface Signals—Spacecraft Propulsion Subsystem	II-34
II-15	Spacecraft Propulsion Subsystem Diagram	II-35
II-16	Selected Spacecraft Propulsion Subsystem	II-39
II-17	Orbit Control Thruster Locations	II-41
II-18	SPS Layout Within EVM	II-42
II-19	Power Subsystem Simplified Block Diagram	II-44
II-20	Power Subsystem Regulation Control Modes	II-44

ILLUSTRATIONS (Cont)

Figure		Page
II-21	Power Subsystem Functional Block Diagram	II-47
II-22	LIC Performance Characteristics	II-51
II-23	Pyrotechnic Functional Diagram	II-53
II-24	Communication Subsystem Block Diagram	II-55
II-25	EVM Layout Showing Communication Subsystem Location	II-57
II-26	Composite Antenna Feed	II-59
II-27	Bottom View of Communication Subsystem Box Showing Transponder and Earth Viewing Horns	II-59
II-28	Transponder Signal Interfaces	II-61
II-29	ATS F&G Telemetry & Command Subsystem	II-70
II-30	Telemetry Subsystem Functional Block Diagram	II-75
II-31	Flexrib Reflector	II-78
II-32	ATS-F Spacecraft Configuration and External Harness	II-81
II-33	Exploded EVM Connector Interfaces	II-81
II-34	EVM Interface	II-82
II-35	Service Module Harnessing	II-82
II-36	Basic Equation Angle Measurement	II-84
II-37	Antenna Array Configuration	II-84
II-38	Block Diagram of Interferometer Subsystem	II-85
II-39	Antenna Array Installation	II-87
III-1	Titan III-C Launch Vehicle	III-2
III-2	Titan III-C Cutaway View	III-3
III-3	Four-Gimbal System	III-8

ILLUSTRATIONS (Cont)

Figure		Page
III-4	Typical Titan III Launch Site	III-8
IV-1	ATSOCC Operational Data Flow	IV-3
IV-2	ATS Communications Landlines	IV-4
V-1	ATS-F and -G Mission Profile	V-2
V-2	ATS-F and -G Nominal Ascent Trajectory	V-2
V-3	Spacecraft Reference Orientation	V-4
V-4	Acquisition Control Modes Block Diagram	V-7
V-5	Operation Control Modes Block Diagram	V-8
VI-1	Rosman Ground Station	VI-2
VI-2	T & C Block Diagram for Rosman & Mojave	VI-4
VI-3	Rosman Ground Station Block Diagram	VI-11
VI-4	Mojave Ground Station	VI-13
VI-5	Mojave Ground Station Block Diagram	VI-15
VI-6	Typical Transportable Ground Station	VI-16
VI-7	T & C Block Diagram for TGS	VI-17
VI-8	TGS Block Diagram	VI-18
VI-9	ATS-F and -G Mobile Terminal	VI-19
VI-10	Vans in Transport Configuration	VI-19
VI-11	ATS-F and -G Mobile Terminal System Block Diagram	VI-22

ILLUSTRATIONS (Cont)

Figure		Page
VII-1	Position Location and Aircraft Communication Experiment System	VII-2
VII-2	860 MHz Antenna Beam Coverage of India for ITV	VII-9
VII-3	TRUST System	VII-11
VII-4	Rosman and Mojave Ground Station Block Diagram	VII-13
VII-5	C-band RFI Experiment	VII-14
VII-6	Visibility Contours	VII-16
VII-7	ATS-F/NIMBUS Data Relay System	VII-19
VII-8	Transmitting Stations Block Diagram	VII-23
VII-9	ATS-F Propagation Experiment System	VII-24
VII-10	Transponder Block Diagram	VII-24
VII-11	Receiving System Block Diagram	VII-25
VII-12	ATS-F Data Relay Experiment	VII-30
VII-13	Polar Motion as a Change in ΔI	VII-34
VII-14	"Footprints" of ETV Coverage in Rocky Mountain States	VII-37
VIII-1	ATS-F and -G/EME Operational Configuration	VIII-4
VIII-2	Environmental Measurements Experiment	VIII-5

TABLES

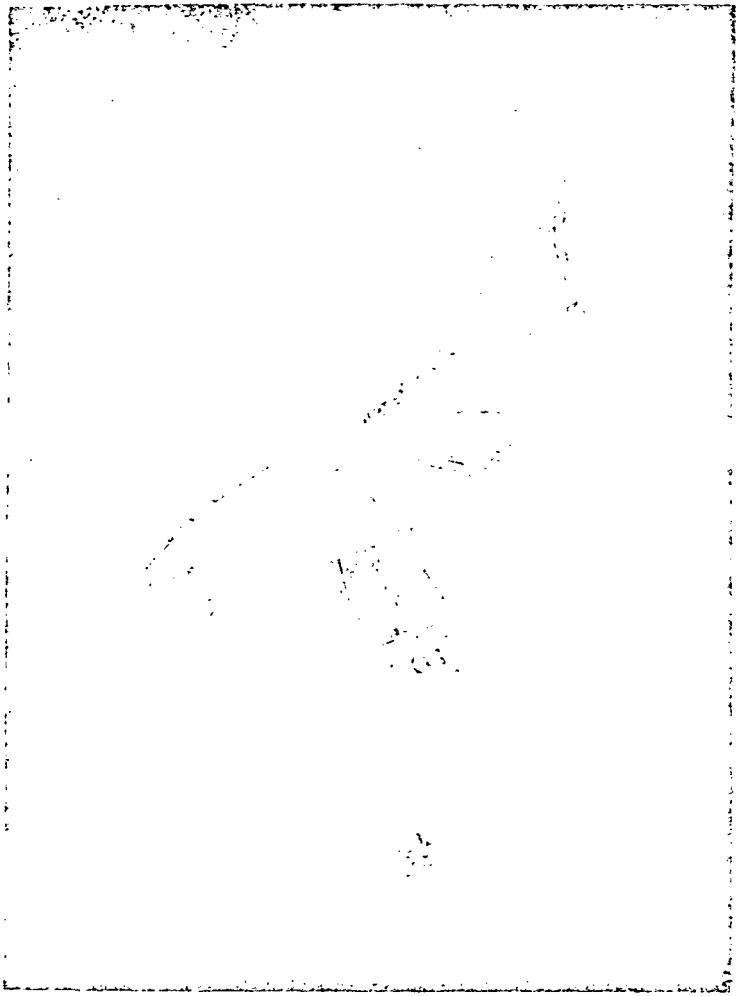
Table		Page
I-1	Applications Objectives Summary	I-2
I-2	ATS-F Mission Profile	I-5
II-1	Summary of Weights	II-9
II-2	*Summary of Mass Properties of Stowed* Configuration	II-11
II-3	Summary of Mass Properties of Deployed Configuration	II-12
II-4	Sensors and *Experiments Field of View*	II-13
II-5	Thermal Requirements	II-14
II-6	Backup Capability—Operational	II-26
II-7	ACS Characteristics Summary	II-27/II-28
II-8	Input Interface Signals—SPS From ACE	II-35
II-9	SPS Mission Requirements	II-42
II-10	Communications Subsystem Characteristics	II-58
II-11	Transponder Signal Interfaces	II-62
II-12	Expected Receive Signal Levels	II-63
II-13	*Frequency Synthesizer Performance* Requirements	II-64
II-14	C-Band Performance	II-65
II-15	S-Band Performance	II-66
II-16	L-Band Performance	II-67
II-17	VHF and UHF Performance	II-68
II-18	*Command Subsystem Characteristics*	II-71
II-19	T & CS Telemetry Parameters	II-76

TABLES (Cont)

Table		Page
II-20	ATS Harness Summary	II-80
II-21	RF Interferometer Subsystem Summary	II-86
II-22	Ground Station to Interferometer Link Parameters	II-88
II-23	Mixer/Amplifier Module Performance Requirements	II-89
II-24	Double Local Oscillator Module Performance Requirements	II-92
II-25	IF Amplifier Detector Module Performance Requirements	II-93
V-1	Backup Capability—Operational	V-9
VI-1	Ground Communications To and From Sites	VI-2
VI-2	ATS Ground Station Capabilities Chart	VI-5
VI-3	T & C Functional Characteristics for Rosman, Mojave, and TGS	VI-10
VI-4	RF Characteristics for the Mobile Terminal	VI-22
VI-5	Mobile Terminal System Parameters	VI-23
VI-6	15-Foot Antenna Subsystem Parameters	VI-23
VI-7	21-Foot Antenna Subsystem Parameters	VI-24
VII-1	PLACE Frequencies (GHz)	VII-5
VII-2	Site Uplink Calculations	VII-7
VII-3	Site Downlink Calculations	VII-8
VII-4	Site Performance Calculations	VII-9

TABLES (Cont)

Table		Page
VII-5	TRUST Uplink Calculations	VII-11
VII-6	TRUST Downlink Calculations	VII-12
VII-7	TRUST Performance Calculations	VII-12
VII-8	Uplink Performance Parameters	VII-27
VII-9	Downlink Performance Parameters	VII-28
VII-10	ETV Experiment Summary	VII-33
VIII-1	Environmental Measurements Experiment	VIII-3



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SECTION I

GENERAL INFORMATION

Introduction

Historical Background

ATS-1 Through -5 Operational Histories

Mission Profile

SECTION I

GENERAL INFORMATION

INTRODUCTION

ATS-F and -G are the latest of a unique series of satellites to be launched into space to conduct approximately fifteen major experiments and a number of related data gathering studies. The series of spacecraft, seven in all, are called the Applications Technology Satellites. The ATS-F and -G are each expected to orbit a minimum of two years.

The Applications Technology Satellites (ATS), one of the largest groups of a single class of satellites to be spaceborne at the same time, are intended to collect and confirm data for various technologies having wide applications in space and space flight.

The underlying design philosophy of the ATS program has been the development of a multiple-mission satellite having a large and adaptable volume for mounting the various experiment payloads, while still satisfying the requirements for spin, gravity-gradient or three axis stabilization.

Of prime importance among the major experiments conducted on board ATS-1 through -5 were those concerned with the stabilization of the satellites by gravity-gradient techniques and providing standarized antenna beams from spinning satellite. The ATS-F and -G spacecraft differ in this respect as they are stabilized by the use of wheels and jets. The multiple-mission design concept of the ATS series allows for numerous different space and earth studies, plus experimentation with advanced satellite communications technologies.

The primary objectives of the ATS-F and -G are as follows (see Table I-1):

- To demonstrate the feasibility of deploying a 30-foot diameter parabolic antenna in space
- To provide a satellite with fine pointing (0.1 degree), slewing (17.5 degrees in 30 minutes) and tracking capability.
- To provide an oriented, stable spacecraft platform at synchronous altitude for advanced technological experiments to be selected by the Space Science and Applications Steering Committee.

Table I-1. Applications Objectives Summary

Experiment	Type of Experiment	Major Objectives
Very High Resolution Radiometer	Meteorological	Measure cloud cover Determine wind field Ocean temperatures/Earth resources
Radio Frequency Interference	Communications	Investigate C-band terrestrial noise sources (geographical/noise power distribution)
LASER Retroreflector	Scientific & Technological	Precise position determination along orbit; redetermine values of GM and geopotential coefficients and investigate time dependence in the geopotential
Millimeter Wave	Communications	Investigation of atmospheric propagation at MMW frequencies. Feasibility of the application of MMW communications systems
Radio Beacon	Scientific	Ionospheric/exospheric electron content measurements. Study of ionospheric propagation effects
Environmental Measurements	Scientific	Particle measurements. Magnetic field measurement. Solar cell degradation.
SAPPSAC	Technological	Investigation of computer controlled attitude control performance optimization
TRUST	Communications	To advance state-of-the-art in space communications by demonstrating CCIR quality wideband signaling between ATS-F and inexpensive ground stations
ITV	Communications	To demonstrate the relay by geo-synchronous satellite of CCIR quality television from a high-powered program transmitting station to small modified standard TV receivers located throughout rural India and to urban rebroadcast stations.
PLACE	Communications	To develop improved air traffic control communications and techniques
TADRSS	Communications	To demonstrate technology necessary for an operational tracking and data relay satellite system
Ion Engine	Technological	Verify and obtain data on Ion Microthruster Propulsion System. Demonstrate thrust vectoring for attitude control.
13- and 15-GHz Propagation	Technological	Collect data on attenuation due to precipitation. Determine power margins needed in spacecraft communications systems.
ETV	Technological	To evaluate a system that will permit relay of programs through the satellite to facilities such as schools, CATV systems, and cities.

The experiments to be performed can be grouped into four basic categories:

- Communications
- Meteorological
- Spacecraft Stabilization
- Scientific

HISTORICAL BACKGROUND

The ATS program evolved directly from the Advanced SYNCOM study program. Its inception was necessitated by rapidly increasing requirements and technological advancements of space systems and concepts in various disciplinary areas. These include spacecraft design, propulsion and stabilization methods and systems; meteorological concepts, applications and techniques; space environmental studies and measurement systems; and communications concepts, techniques and applications.

The ATS program began with the termination of the Advanced SYNCOM program. The Advanced SYNCOM engineering design studies provided much of the base for the ATS. SYNCOM program techniques, especially for synchronous altitude, spin-stabilized configurations and complex synchronous orbit maneuvers, were adopted for the synchronous altitude spin-stabilized flights.

Prior to the ATS program, several spacecraft programs were being planned by COMSAT and NASA that were nearly identical to the Advanced SYNCOM project.

The huge cost reduction and efficiency to be achieved in consolidating these multiple efforts into a single program became readily apparent, and so the ATS program using the Advanced SYNCOM as a base was established.

The ATS series of spacecraft were an outgrowth of previous communications satellites. These predecessors and their contributions to the ATS program are outlined below.

Satellite/ Department	Date Launched	Contribution
SCORE-DOD	Dec. 1958	Recorded message from President to world.
COURIER-DOD	1960	Delayed repeater - investigated store-forward and real-time capabilities.
WESTFORD-DOD		Tiny resonant dipoles dispersed in orbital belt for communication by reflection.

Satellite/ Department	Date Launched	Contribution
ECHO I - NASA	1960	Passive communications using inflatable sphere of aluminized mylar.
ECHO II - NASA	1964	Same as above, plus collaboration with Russia.
TELSTAR - AT&T - BTL	1962	Active wideband communications. Launch and early orbit determination done by NASA.
TELSTAR II - AT&T - BTL	1963	Same as above.
RELAY I, RELAY II - NASA	1962 & 1964	TV and voice experiments between U.S., Europe, Brazil, and Japan.
SYNCOM I - NASA & DOD	1963	During apogee injection, contact was lost.
SYNCOM II - NASA & DOD	1963	Demonstrated stationkeeping and orbital control principles. International communications. First use of Goddard range and range-rate.
SYNCOM III - NASA & DOD	1964	First stationary orbit. Wide and narrow band repeater for all modes of communications between Far East and the U.S.
EARLYBIRD - COMSAT	1965	Operational communications. Launch and early orbit determination done by NASA.
TIROS I - NASA	1960	Monitored global weather conditions and cloud cover. Predecessor of NOAA series.
NIMBUS - NASA	1964	Improved over TIROS, incorporated earth stabilization & modular construction.

The larger payload capacity ATS spacecraft expands the capabilities of Advanced SYNCOM but incorporates many of its concepts into a highly refined and integrated multi-functional package.

ATS 1 THROUGH 5 OPERATIONAL HISTORIES

The long list of achievements and accomplishments of the ATS program increases almost daily. With the exception of ATS-2 and -4, as noted below, each system and every experiment of the ATS satellites orbited to date has functioned as designed and sent back to earth invaluable information, verifying the technology, and leading to better and wider applications in the future.

ATS-1 (ATS-B before launch)

This spin-stabilized, synchronous-altitude satellite was the first ATS to be launched. Soon after its lift-off on December 7, 1966, this satellite was maneuvered into its permanent position over the Pacific Ocean and all its systems checked out as fully operational. Since then, all the experiments have been pouring a steady stream of data back to the ground stations on earth.

Some services performed by ATS-1 in addition to technological and scientific experiments were the relay of special TV programs such as the first splashdown of the APOLLO moonflight, EXPO 67 in Australia, and the Japanese Prime Minister's visit to Australia; the performance of special SHF tests such as mass data transfer and antenna beam control during the eclipse; relay of special VHF communications such as emergency communications during the Alaskan flood and magnetic studies at field line base; and relay of special weather facsimile for air operations in the Pacific. ATS-1 also gave real-time data support to the ground stations operating with INTELSAT II and to OGO-IV and OSO-IV satellites.

ATS-2 (ATS-A before launch)

The second satellite in the ATS series was launched on April 6, 1967. It was intended that this satellite be placed in a medium altitude orbit inclined to the earth's equator, so that it would continuously circle the earth. Unfortunately, a failure in the fuel supply system of the Agena rocket prevented the booster from reigniting at the crucial point in the satellite trajectory, and the satellite went into a highly elliptical orbit around the earth. Large forces, caused by the severe decelerations occurring in this very irregular orbit, have been more than the gravity-gradient stabilization system can overcome, and as a result, ATS-2 slowly tumbled and rotated in orbit. However, again

all the systems and experiments on board functioned as well as possible under these conditions. The spacecraft reentered the earth's atmosphere on September 2, 1969 and was destroyed.

ATS-3 (ATS-C before launch)

The third satellite in the series, a spin-stabilized spacecraft, was put into its synchronous altitude orbit on November 5, 1967. After a perfect launch, it took up its station over the Atlantic Ocean, and all systems have been operating as expected since then.

One of the most notable of the achievements of this satellite is the first ground-to-satellite-to-airplane communications link over the Atlantic Ocean. This historic event took place on November 21, 1967, during a Pan American flight from New York to London. The communications engineer aboard the plane was able to maintain conversations with both Pan American officials in New York and with NASA officials at Goddard Space Flight Center. The Mojave ground station actually handled the experiment control, and transmissions to and from the aircraft via the satellite were monitored at different places around the world as far away as London, Hamburg, Frankfurt, and Buenos Aires.

A close competition for the most notable achievement of ATS-3 was the color photograph obtained from the Multicolor Spin-Scan Cloud Camera.

The Multicolor Spin-Scan Cloud Camera was not the only camera on ATS-3, there was also the Image Dissector Camera (IDCS). The IDCS obtained many cloud photographs.

ATS-3 performed many services in addition to the technological and scientific experiments. Some of these services included support of the Apollo missions; TV coverage of Pope Paul's visit to Bogota, Columbia; special relay tests such as the IEEE International Communication Conference; VHF relay of special commands to ATS-4 from Johannesburg; demonstration of automatic picture transmission at NASA headquarters by the weather facsimile (WEFAX) experiment on board the ATS-3; real-time data support of the OSO-IV satellite, and provided an RF collimation source for the Canadians.

ATS-4 (ATS-D before launch)

ATS-4 was launched from Cape Kennedy on August 10, 1968. The Centaur second ignition was never accomplished and, as a result, the separation of the spacecraft from the Centaur was also not accomplished. The spacecraft, with

Centaur and apogee motor attached was left in an approximately 100 by 400 nautical mile orbit. Turn-on of all systems and experiments was accomplished; however, little useful data were obtained because of the extreme anomalistic conditions. One thing that was proved, however, was the feasiblity of erecting a large body through the use of nutation. ATS-4 reentered the earth's atmosphere and was destroyed on October 17, 1968.

ATS-5 (ATS-E before launch)

ATS-5 was launched from Cape Kennedy on August 12, 1969 and has contributed much in the way of technical and scientific information.

The Lockheed auroral particles experiment on ATS-5 has now been operating successfully since 1969 and is continuing to provide a large quantity of useful and interesting data.

The NASA/GSFC L-band ranging and position location experiments, being conducted from the Mojave ground station to the ATS-5 satellite, have demonstrated the ability to obtain meaningful range measurements using PM tone modulation at L-band carrier frequencies. The objective of this experiment is to demonstrate the capability of obtaining ranging measurements between an earth station and the ATS-5 satellite at L-band and C-band carrier frequencies, and to evaluate the relative accuracy to which these measurements can be made.

Applied Information Industries (AII) conducted the Alpha II Project for SAMSO, U.S. Air Force, by doing simultaneous L- and C-band ranging using ATS-5. AII also performed a ranging test for the Maritime Administration using the tanker, USS Manhatten. This test was basically a ranging test to determine line-of-position of the ship. L-band ranging signals were transmitted from Mojave ground station and relayed through ATS-5 to the ship.

The MARAD project (Maritime Administration) was also conducted by AII. This was a successful test of real-time high speed (100 wpm) teletype (TTY) using standard equipment and relayed through ATS-5, two-way (not simultaneous), from Mojave to the Manhatten. The TTY aboard the ship was unattended and running continuously.

Among the other tests and experiments that have been conducted are the Ionospheric Propagation tests, an on-going experiment involving three grounds stations at Quito, Lima, and Mojave that accepted fading and fading rate data from ATS-5 on L-band and VHF simultaneously; the Federal Aeronautics Administration/Boeing Aircraft Company experiment at L-band from ATS-5 to an aircraft involving measurements of multipath effects and tone ranging; the

Millimeter Wave experiment conducted by Westinghouse for GSFC concerning the propagation correlation with rain, fading, and weather conditions at 31.65 GHz uplink and 15.3 GHz downlink; and the All/Wallops Island experiment in relative position (ranging).

MISSION PROFILE

The ATS-F Mission Profile (Table I-2) can be divided into ten time phases as described in the following paragraphs.

Table I-2.
ATS-F Mission Profile

TIME		MISSION PHASE	
Days	Hours		
0	0	I.	The launch into synchronous orbit
0	6.5	II.	Deployment and acquisition into a local vertical orientation
0	7.7	III.	Orbit determination and begin-orbit correction
0	8.0	IV.	Begin spacecraft performance tests
7	0	V.	End orbit correction maneuver and command the spacecraft into a reference orientation
14	0	VI.	Completion of spacecraft performance tests
14	0	VII.	Begin operations at 94° W
365	0	VIII.	Eastward repositioning to 35° E
405	0	IX.	Begin operations at 35° E
770	0	X.	Westward repositioning to 94° W

Phase I-Launch into Synchronous Orbit—The spacecraft will be launched from the Eastern Test Range (GSFC) using a Titan III-C launch vehicle at a launch azimuth. At the second equatorial crossing (over the Mid-Pacific)

the spacecraft is injected into a transfer orbit. It achieves synchronous altitude about 5 1/4 hours later (about 6 1/2 hours after launch). At this time Stage III (transtage) reignites to put the spacecraft into a synchronous equatorial circular orbit between 93.3° to 95°West longitude. The spacecraft is then separated from the transtage.

Phase II - Deployment and Acquisition into a Local Vertical Orientation—After nulling out any angular rates imparted to the spacecraft during separation, the solar panels and then the 30-foot parabolic reflector are deployed. After deployment, commands are sent to the various sensors to acquire sequentially the sun, earth, and Polaris. The sun sensor is used for initial orientation. The attitude control subsystem uses the earth sensor output to stabilize the roll and pitch axes and the Polaris sensor to stabilize the yaw axis. Figure I-1 depicts the launch separation and acquisition sequence.

Phase III - Orbit Determination and Correction—After about four hours, the ATS-F orbital parameters will have been accurately determined. Based on these data appropriate commands will be transmitted to the spacecraft from the ground to make the necessary corrections. Because of the limited thrust available this maneuver may take as much as one week.

Phase IV - Spacecraft Performance Tests and Orbit Correction—Up to two weeks will be required to check out the performance of the spacecraft. Certain tests can be performed during the time when orbit correction is being made.

Phase V - End of Orbit Correction Maneuver and Placement of Spacecraft in Operational Reference Attitude—When the spacecraft has arrived at its correct location and its orbit verified, the attitude control subsystem will be commanded to the reference orientation mode (+Z, yaw, to local vertical; +Y, pitch, normal to orbit plane, toward south pole; +X, roll, in orbit plane, in normal direction of motion).

Phase VI - Completion of Spacecraft Performance Tests—All tests of the spacecraft subsystems and the experiments that require movement of the spacecraft will be conducted at this time. After all systems and experiments have been checked out, the spacecraft will be declared operational.

Phase VII - Operations at 94° West—Most experiments will be performed at this location. This phase will last approximately one year. The prime ground stations for operation at this location are Rosman and Mojave.

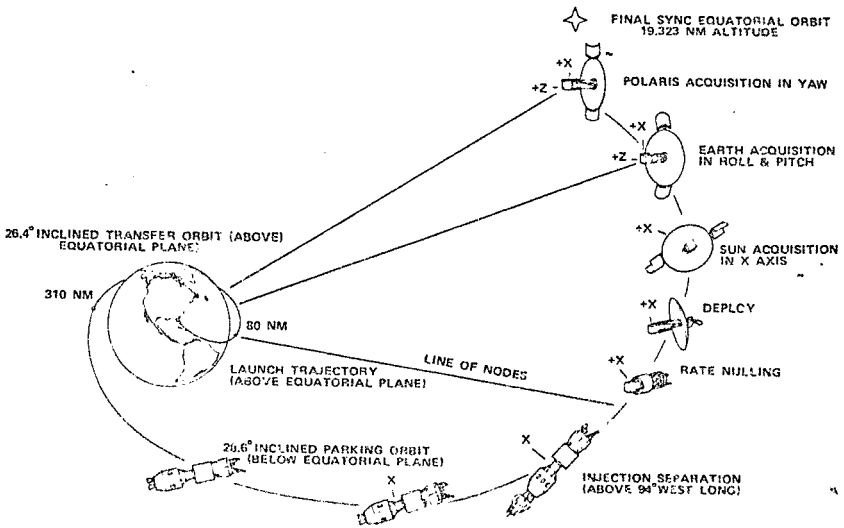


Figure I-1. ATS F and G Launch and Acquisition Sequence

Phase VIII - Eastward Repositioning to 35° East—At the conclusion of the testing period over the United States, the spacecraft will be repositioned to a point at 35° East longitude over Africa. This maneuver will take approximately 41 days.

Phase IX - Operations at 35° East—Operation of the satellite will be similar to that for 94° West except that the prime ground station will be the Transportable Ground Station (TGS) in Europe.

Phase X - After One Year of Operation at 85° East—S/C will be moved to 94° W to conduct additional experiments as required.

SECTION II

SPACECRAFT

Structure

Thermal Control Subsystem

Attitude Control Subsystem

Spacecraft Propulsion Subsystem

Electrical Power Subsystem

Communications Subsystem

Telemetry and Command Subsystem

Reflector

Harness

RF Interferometer Subsystem

SECTION II

SPACECRAFT

STRUCTURE

The ATS-F and -G spacecraft is shown in orbiting configuration in Figure II-1. The functional requirements of the spacecraft dictate the need for a high degree of accuracy in the mechanical alignment and orientation of the various experiments and structural elements. The spacecraft consists of five major structural elements (Figure II-2).

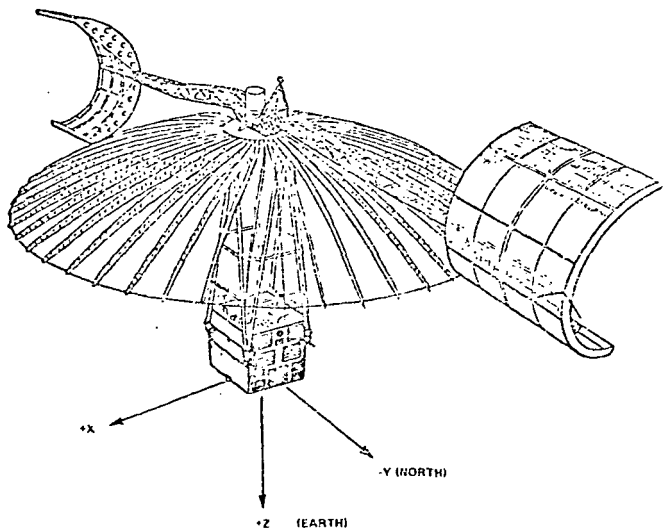


Figure II-1. ATS-F Configuration in Orbit

- Earth Viewing Module (EVM)—The EVM basic structure is a cube 54 inches on a side by 65.45 inches high made up of three subsections. The lower subsection is designated the experiment module. It contains most of the experiments and all of the earth-viewing

sensors. The middle section of the EVM is called the service module. It contains the components of the electrical power subsystem, the telemetry and command subsystem, and the attitude control subsystem. The topmost section is the communications module. The module contains the transponder and three experiments closely associated with the communications subsystem. The prime focus feed for the high gain antenna system is mounted on the top surface of the EVM.

- Structural Hub—The structural hub (Figure II-3) is an open square 42 inches on a side and 6 inches deep. The diagonals of the square lie along the X and Y axes of the spacecraft. The hub, at its four corners, supports the 30-foot diameter reflector. It also provides a mounting surface for the solar array trusses and for the experiments which view deep space.

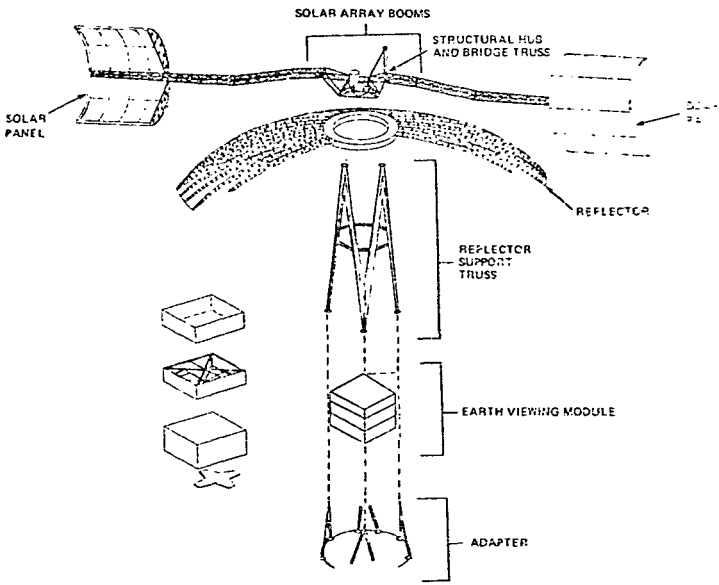


Figure II-2. Spacecraft Structural Assemblies

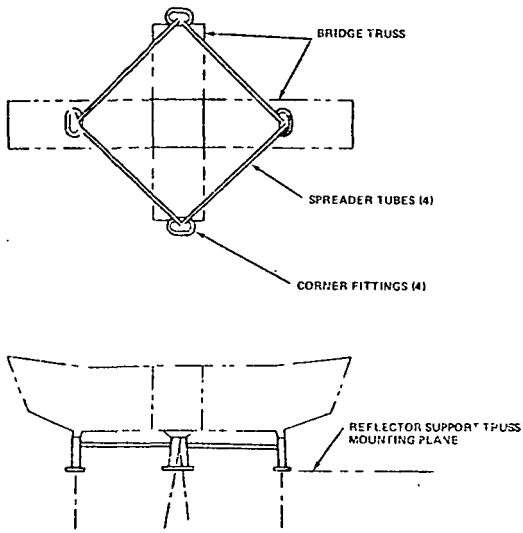


Figure II-3. Structural Hub

- Reflector—The reflector is a 30-foot diameter parabolic reflector consisting of flexible aluminum radial ribs supporting mesh that forms a paraboloidal surface. It has a focal length-to-diameter (F/D) ratio of 0.44. During launch the reflector is stowed in a toroidal-shaped container with an outside diameter of about 75 inches. When deployed, the ribs unfurl to form the parabolic shape.

- Reflector Support Truss—The EVM and the structural hub are interconnected by the reflector support truss (Figure II-2), which consists of eight tubular members arranged in a square symmetrical A-frame configuration. The top ends are joined to the four corners of the structural hub. The bottom ends are joined to the adapter and to the EVM at the midpoints of each of the four vertical edges of the EVM.

- Solar Array—The solar array (Figure II-2) is made up of two semi-cylindrical panels, each having an approximate radius of 54 inches and a length of 94.6 inches. Each panel is attached to the end of a support boom that is mounted to the structural hub (Figure II-4). Solar cells are located on the convex surface of the curved honeycomb sandwich panels.

The deployed spacecraft (Figure II-5) spans about 52 feet from the ends of the solar arrays, 30 feet across the parabolic reflector, and about 25 feet from the earth-viewing surface of the EVM to the top of the magnetometer. Initial orbital weight is approximately 2000 pounds.

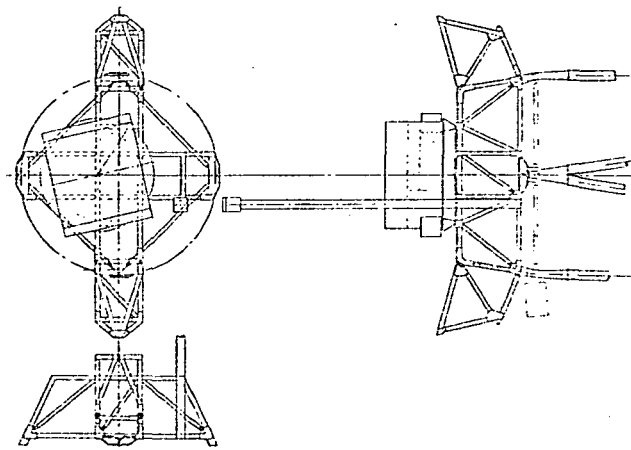


Figure II-4. Hub and Bridge Truss

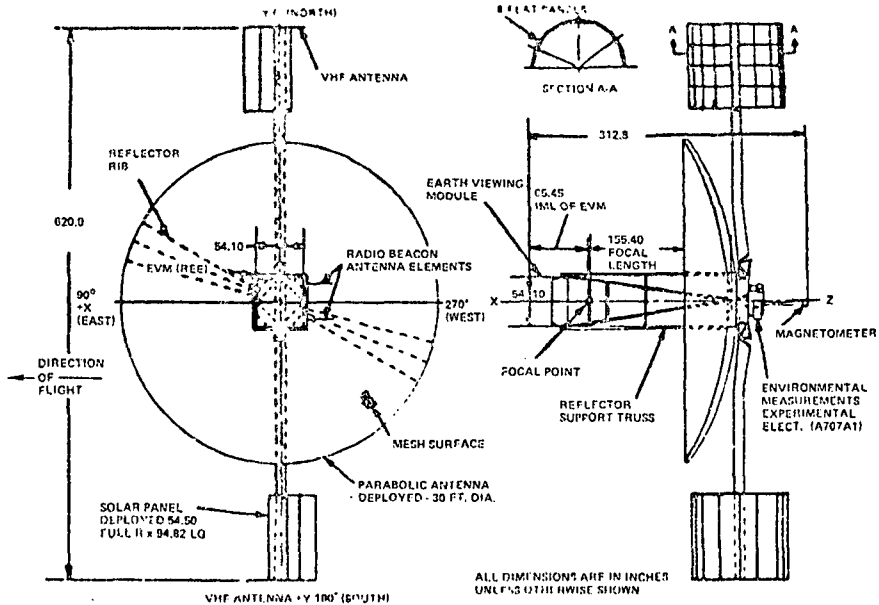


Figure II 5. Spacecraft Deployed Configuration

Launch Configuration—For launch, the reflector is stowed in the torus around the structural hub while the solar array booms are folded down at the knee joints that interconnect the booms to the truss structure of the structural hub. (See Figure II-6.) The two semicylindrical solar panels are folded back over the booms to form a complete cylinder with its lower edge approximately 3 inches above the prime focus feed surface of the EVM.

The EVM is mounted to the Titan III-C transtage and is located about 5 inches above the transtage by an adapter consisting of four independent inverted-V tubular trusses. The vertices of the V's are attached to the EVM at approximately the midpoint of the four vertical edges. The open ends of the V's are attached to eight points on the Titan III-C field joint.

Deployment—When orbital position has been reached and the spacecraft is in an inertially stable condition, separation of the spacecraft and transtage is initiated by a signal from the transtage which fires the pyrotechnic separation attachments on both the transtage and spacecraft sides of the separation plane. Separation is verified by a signal and transmitted through the spacecraft telemetry systems. In the event of failure, a backup is provided through the spacecraft command systems.

After separation and stabilization of the spacecraft, upon command, the solar panel constraints are released and the solar array support booms deploy about the simple knee joints at the structural hub truss (Figure II-7). Following lock-up of these joints, upon command, the solar panels rotate about the skewed hinges, assuming positions above and outboard of the reflector deployment envelope. The reflector is then deployed upon command from the ground, after which the solar array booms rotate downward about 22 degrees and lock in this orbital position.

Additional information is provided in the following tables:

II-1. Summary of Weights
II-2. Summary of Mass Properties of Stowed Configuration
II-3. Summary of Mass Properties of Deployed Configuration
II-4. Sensors and Experiments Field of View

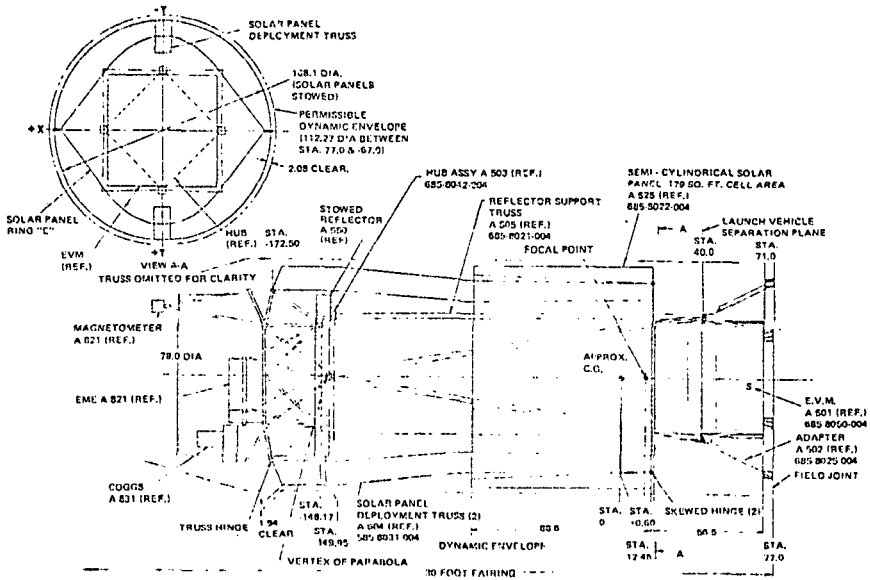


Figure II-6. Spacecraft Launch Configuration

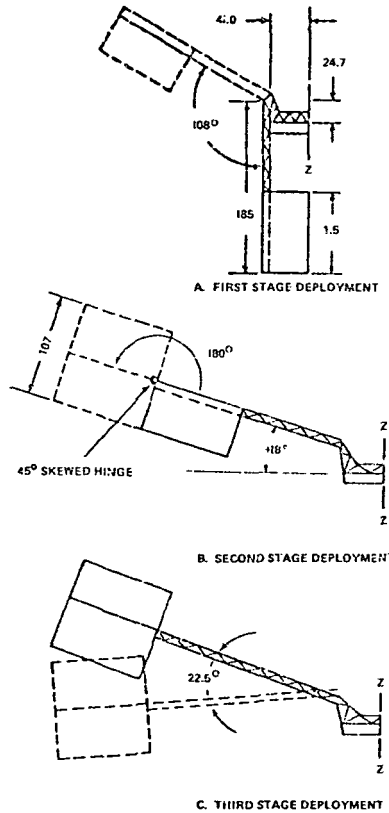


Figure II-7. Solar Panel Deployment

Table II-1
Summary of Weights

Description	Target Weight	August 1971			July 1971	June 1971
		Minimum	Nominal	Maximum		
Structures	(525)	(561)	(613)	(635)	(616)	(601)
Hub Structure		65	94	77	78	72.6
EVM Structure		286	267	301	255	258.4
Reflector & Support Truss		77	78	80	78	78.0
Adapter & Separation		56	56	59	59	58.6
Deployment Booms		79	118	118	113	103.6
Attitude Control	(190)	(216)	(226.5)	(239)	(225.2)	(225.2)
ACS Sensors		63	66.4	70	67	67
ACS Controllers		57	61.9	66	63	63
ACS Wheels		58.5	58.5	59	58.5	58.5
ACE		19.5	20.6	22	21	21
Interferometer		16	19.1	22	15.7	15.7
Spacecraft Propulsion	(140)	(158)	(159.4)	(177)	(160.9)	(160.9)
SPS Hardware		57	57.4	67	58.9	58.9
Fuel/Pressurant		101	102	110	102	102
Telemetry & Command	(95)	(91)	(99.1)	(146.5)	(98.1)	(101.5)
RF & DSU		30	30	53	30	31.5
DACU		27	33.6	33.5	33.6	33.5
CDD		34	34.5	60	34.5	60
Communications	(240)	(275)	(301)	(340)	(301)	(301.0)
Transponder		207	236	245	233	233.0
Prime Focus Feed		60	59	83	60	61.0
Radio Beacon		8	8	12	8	8.0
Reflector	(150)	(146.5)	(162.1)	(171)	(163)	(163.1)
Ribs		97.2	97.3	103	97	97.2
Hubs		40.3	54.9	12	55	54.9
Mesh		9	8.6	56	9	11.0
Electrical			1.3		2	
Electrical Power	(311.0)	(299)	(343)	(351)	(344.1)	(342.6)
Solar Panels		170	200	190	159.1	154.1
Batteries		67	72	77	75.0	77.0
PRU & Shunts		34	38	50	47.0	47.0
Electronics		28	33	34	33.0	33.5
Wiring Harness	(140.0)	(135)	(150.5)	(175)	(152.5)	(157.5)
External		34	46.7	47	46.7	46.7
Internal		101	103.8	128	105.8	110.8
Thermal Control	(140)	(147)	(163)	(175)	(163)	(160.0)
Heat Pipes & Installation		64	66	68	66	66.0
Louvers		38	39	47	32	34.0
Passive Protection		45	58	60	55	56.0

Table II-1
Summary of Weights (Cont)

Description	Target Weight	August 1971			July 1971	June 1971
		Minimum	Nominal	Maximum		
Experiments	(450)	(512)	(528.4)	(539)	(519)	(495)
EME		145	145	145	145	145.0
M/M Wave		90	90	95	90	86.3
Laser Reflector		40	40	45	40	41.1
Radiometer		88	90	92	90	85.1
Ion Engine		67	68	70	68	70.0
Propagation		40	40	42	40	43.0
			7			
Experiment Support			5			
Structure		20	22	24	22	22.6
LIC's		16	15.4	19	15	14.2
Instrumentation		6	6	6	6	
TOTAL ATS		2,540.5	2,750.4	2,947.5	2,745.[illegible]	2,[illegible]

Table II-2
Summary of Mass Properties of Stowed Configuration

1. Total spacecraft weight 2005.8 pounds

2. Center of mass of satellite

   X = -0.84 inches; Y = 0.67 inches; Z = -76.40 inches (Spacecraft station -0.60)

3. Moments of inertia about axes through center of mass parallel to reference axes.

I_{XX} = 2819.3 slug-ft^2	I_{XY} = 16.1 slug-ft^2	I_{XZ} = 26.0 slug-ft^2
I_{YX} = 16.1 slug-ft^2	I_{YY} = 2755.6 slug-ft^2	I_{YZ} = -4.4 slug-ft^2
I_{ZX} = 26.0 slug-ft^2	I_{ZY} = -4.4 slug-ft^2	I_{ZZ} = 381.3 slug-ft^2

4. Direction cosines of principal axes relative to axes in Item No. 3 above

	X_{REF}	Y_{REF}	Z_{REF}
X_P	.97311	.23014	.00997
Y_P	-.23011	.97316	-.00432
Z_P	-.01069	.00191	.99994

5. *Principal moments of inertia about principal axes in Item No. 4 above*

   I_{XP} = 2823.4 slug-ft^2

   I_{YP} = 2751.7 slug-ft^2

   I_{ZP} = 381.0 slug-ft^2

Table II-3
Summary of Mass Properties of Deployed Configuration

1. Total spacecraft weight 1956.4 pounds

2. Center of mass of satellite

   X = -0.86 inches; Y = 0.69 inches; Z = -92.26 inches (Spacecraft station -15.26)

3. Moments of inertia about axes through center of mass parallel to reference axes

I_{XX} = 7157.0 slug-ft^2	I_{XY} = -177.2 slug-ft^2	I_{XZ} = 31.8 slug-ft^2
I_{YX} = -177.2 slug-ft^2	I_{YY} = 3939.5 slug-ft^2	I_{YZ} = -9.0 slug-ft^2
I_{ZX} = 31.8 slug-ft^2	I_{ZY} = -9.0 slug-ft^2	I_{ZZ} = 3857.9 slug-ft^2

4. Direction cosines of principal axes relative to axes in Item No. 3 above

	X_{REF}	Y_{REF}	Z_{REF}
X_P	.99845	-.05484	.00974
Y_P	.05554	.99359	-.09843
Z_P	-.00428	.09881	.99510

5. Principal moments of inertia about principal axes in Item No. 4 above

   I_{XP} = 7167.2 slug-ft^2

   I_{YP} = 3930.4 slug-ft^2

   I_{ZP} = 3857.1 slug-ft^2

Table II-4
Sensors and Experiments Field of View

Experiments	FOV Coverage	View Direction
VHRR	20° cone	Toward Earth
MMW (AN-1)	15° cone	Toward Earth
Radio Beacon	15° to 60° cone	Toward Earth
EME		
RFI	None	
Radiometer	±45°	South
S/C Components		
Polaris Sensor	±12.5° z axis ±4.5° y axis	North
Earth Sensor	45.5° x 45.5°	Toward Earth
Coarse Sun Sensor	Spherical (by overlapping)	
Fine Sun Sensor	±15°	East-West
Target Suns Sensor (part of fine sun sensor)	± 9°	East-West

THERMAL CONTROL SUBSYSTEM

Because of the large power capability of the ATS-F and -G satellites and the variety of operating modes, there is a wide fluctuation in the amount of power that must be dissipated from the components within the EVM. The thermal control subsystem maintains the temperature of the mounting surface in the EVM within a total temperature excursion of 30° C. It also minimizes temperature variations that would tend to degrade the alignment accuracy of the spacecraft subsystems. See Table II-5 for the thermal requirements of the thermal control subsystem.

Subsystem Design

The thermal control subsystem makes use of superinsulation, thermal louvers, heat pipes, thermal coatings, cubical spacecraft configuration, and heat dissipation components to achieve the objectives described above.

Table II-5
Thermal Requirements

In-Orbit Operation	
Batteries	0°C to 25°C
Component Mounting Surfaces Within EVM	5°C to 35°C
Solar Paddles	-140°C to 50°C
Solar Paddle Support Arms	(-160°C to 50°C)*
GFRP Truss	(-160°C to 45°C)*
Squibs, etc.	(-75°C to 70°C)*
Antenna Hub	(-160°C to 35°C)*
Magnetometer Support	(-75°C to 38°C)*
EME Support	(-75°C to 38°C)*
Earth Sensor	10°C to 30°C
Polaris Sensor	10°C to 30°C
IRA	10°C to 30°C
Thruster Valves on Truss	5°C to 35°C
Propellant Lines on Truss	5°C to 35°C
Thruster Catalyst Beds	(65°C to 750°C)*

Prelaunch, Ascent, Parking and Transfer Orbit Modes	
Batteries	0°C to 25°C
Component Mounting Surfaces Within EVM	5°C to 35°C
Solar Paddles	-140°C to 50°C
Solar Paddle Support Arms	(-160°C to 35°C)*
GFRP Truss	(-160°C to 45°C)*
Squibs, etc.	(-75°C to 70°C)*
Antenna Hub	(-160°C to 35°C)*
Magnetometer Support	(-75°C to 38°C)*
EME Support	(-75°C to 38°C)*
Earth Sensor	5°C to 30°C
Polaris Sensor	5°C to 30°C
IRA	5°C to 30°C
Thruster Valves on Truss	5°C to 35°C
Propellant Lines on Truss	5°C to 35°C
Thruster Catalyst Beds	(65°C to 750°C)*

Acceptance Test Temperature Limits	
Batteries	0°C to 35°C**
Component Mounting Surface Within EVM	5°C to 35°C
Solar Paddles	-140°C to 50°C
Solar Paddle Support Arms	(-160°C to 50°C)*
GFRP Truss	(-160°C to 45°C)*
Squibs, etc.	-75°C to 70°C)*
Antenna Hub	(-160°C to 35°C)*
Magnetometer Support	(-75°C to 38°C)*
EME Support	(-75°C to 38°C)*
Earth Sensor	10°C to 30°C
Polaris Sensor	10°C to 30°C
IRA	10°C to 30°C
Thruster Valves on Truss	5°C to 35°C
Propellant Lines on Truss	5°C to 35°C
Thruster Catalyst Beds	(65°C to 750°C)*

Qualification Test Temperature Limits	
Batteries	-10°C to 40°C
Component Mounting Surfaces Within EVM	-10°C to 50°C
Solar Paddles	-155°C to 75°C
GFRP Truss	(-185°C to 60°C)*
Squibs, etc.	(-90°C to 85°C)*
Earth Sensor	-5°C to 45°C
Polaris Sensor	-5°C to 45°C
IRA	-5°C to 45°C
Thruster Valves on Truss	5°C to 50°C
Propellant Lines on Truss	5°C to 50°C
Thruster Catalyst Beds	(750°C to 785°C)*

- Superinsulation—The superinsulation covers all but the north and south side of the EVM. It consists of 30 layers of 1/4-mil aluminized Mylar with a 5-mil aluminized Kapton coversheet. This combination of a high infrared reflectance material and a low conductance material produce a superinsulation blanket with an effective emittance that does not exceed 0.015.

- Thermal Louver—The north and south faces of the EVM are honeycomb panels containing 17 square feet of thermal louver arrays (Figure II-8). These arrays are rigid, lightweight, polished aluminum frames containing low-friction pivots for the louver blades. The blades are thin, highly polished, specular finish aluminum. They are actuated by bimetallic sensors located in a housing thermally insulated from the exterior environment. When the interior temperature rises, causing the bimetallic sensors to expand, the louver

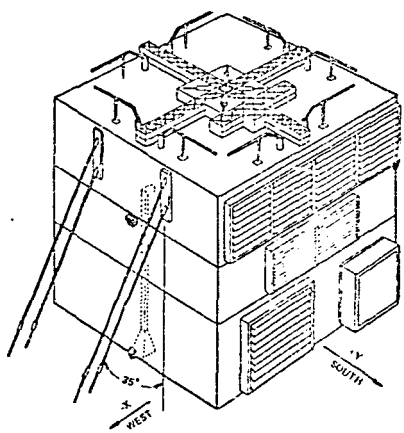


Figure II-8. Earth Viewing Module Showing Thermal Louvers

blades open and allow more heat to be radiated into space. A temperature change of 25°F will cause the louvers to rotate from fully open to fully closed. The surfaces behind the louvers are covered with OSR (Optical Solar Reflector), a vapor deposited silver on an 8-mil thick fused silica. This coating provides a low absorbtance (α) and a high emissivity (ϵ) and has a low level of degradation in the space environment.

- Heat Pipes—Heat pipes fabricated from 1/2-inch square aluminum tubes are bonded as an integral part of the north and south skin panels and of the transverse bulkhead connecting the north and south panels (Figure II-9). In all cases the heat pipes run parallel to the top and bottom surface of the EVM. Heat pipes have a high heat transport capability and maintain near isothermal conditions along their length. Ammonia is used as the working fluid because it has a high heat transfer capability and is compatible with aluminum.

- Thermal Coatings—To minimize the temperature rise of the solar array, the space-viewing side of the substrate is coated with Thermatrol paint (α= 0.20 and ϵ = 0.85).

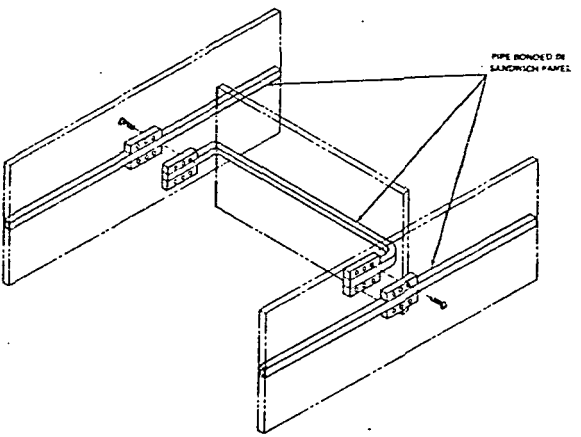


Figure II-9. Typical Heat Pipe Installation

Thermal control of the reflector support truss is attained by covering the truss tubes with aluminized Kapton ($\alpha = 0.4$ and $\epsilon = 0.8$), keeping the Kapton surface on the outside. The truss material, Graphite Fiber Reinforced Plastic (GFRP), has a coefficient of thermal expansion on the order of 10^{-7} inch/inch/°F; therefore, the thermal distortions of the reflector truss are extremely small.

ATTITUDE CONTROL SUBSYSTEM

To perform many of the experiments on the ATS-F and -C, it is essential that the spacecraft be maintained in a precisely controlled stable attitude in orbit for some experiments or be slewed in an accurately prescribed pattern for others. The attitude control subsystem (ACS) provides the necessary attitude stabilization and accurate slewing control; thus, it plays an important part in the successful completion of many of the technological experiments. A block diagram of the ACS is shown in Figure II-10.

The ACS stabilizes and orients the spacecraft in three axes after it has separated from the launch vehicle. The ACS uses radiant energy from the earth and light from Polaris for the prime attitude control references, the DOC as its prime controller, three interia wheels, and/or the attitude control jets of the Spacecraft Propulsion Subsystem (SPS) as its control actuators. It receives commands from the Command Decoder and Distribution Unit (CDDU) to perform various functions associated with onboard experiments and with ACS performance.

The ACS uses the attitude control jets of the SPS to unload the momentum wheels, to maintain attitude control during the orbit control mode, and for a minimum period of 6 months if the inertia wheels fail (jet only mode). Orbit control commands from the ground are relayed to the orbit control jets of the SPS. The ACS also generates and conditions status and monitoring signals for telemetering to the ground via the Data Acquisition and Control Unit (DACU) and receives ground commands to control wheels and jets via the Ground Attitude Control (GAC) decoder as well as the command decode and distribution (CDD) units.

The modes of operation in which the ACS is required to perform may be grouped in two categories, with a number of modes in each:

- Acquisition modes

  Rate damping
  Sun acquisition
  Earth acquisition
  Polaris acquisition

- Operational and experimental modes

  Reference orientation (local vertical)
  Pointing (offset from local vertical)

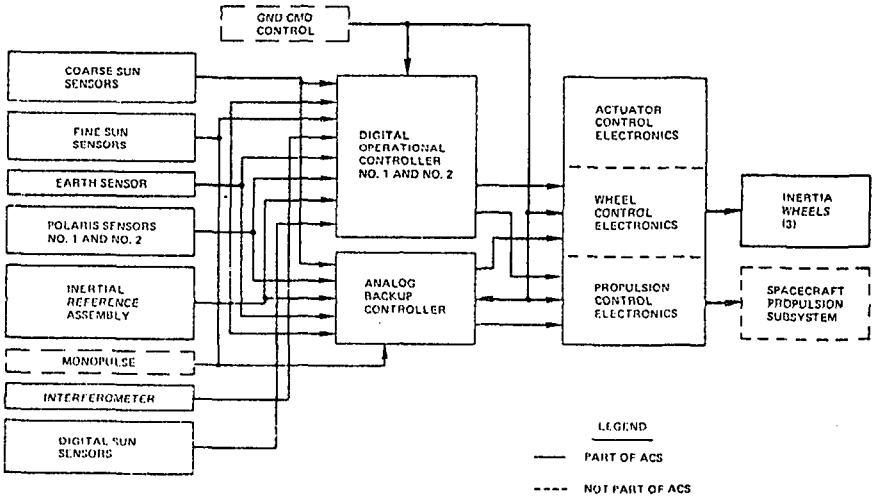


Figure II-10. Simplified ACS Block Diagram

Slew/antenna pattern maneuvers
Station nulling (interferometer and monopulse)
Satellite tracking
Low jitter
SAPPSAC (self-adaptive precision pointing spacecraft attitude control)
Orbit control (repositioning and stationkeeping)

Functions performed by the ACS include:

- Acquisition of the reference local vertical, orbit-plane orientation
- Precision pointing anywhere on the earth's disc up to ±10.0 degrees from local vertical
- Generation and execution of low altitude satellite tracking commands anywhere within ±11.0 degrees from the local vertical
- Generation and execution of slew and antenna pattern measurement maneuver command within ±5 degrees from the line-of-sight to the monitoring ground station in pitch and roll, nominally ±11 degrees off the local vertical
- Provision of a stable earth-referenced platform with low jitter amplitude and jitter rate.

The primary logic functions of the ACS (Figure II-11) are performed by the Digital Operational Controller (DOC) or the Analog Backup Controller (ABC). The primary set of sensors are the earth sensor for roll and pitch and the Polaris sensor for yaw. Other sensors are: the Inertial Reference Assembly (IRA), the coarse and fine sun sensors, the digital sun sensor, and the VHF monopulse. Backup sensors are available for each of the operational modes. The Actuator Control Electronics (ACE) and the three inertia wheels make up the remainder of the subsystem. The components required to effect the thrust are considered part of the Spacecraft Propulsion Subsystem.

Subsystem Description

The ACS consists of a number of major components integrated into functional assemblies. These are shown in Figure II-11-A, -B and -C. In addition to interfacing with these onboard components, the ACS also interfaces with the VHF, C-band and S-band monopulse for input roll- and pitch-attitude error signals.

Digital Operational Controller—Two digital operational controllers (DOC) are provided in the ACS. One DOC serves as the primary controller for all modes of operation and the other is used as a backup. Either DOC may be chosen by command from the ground station.

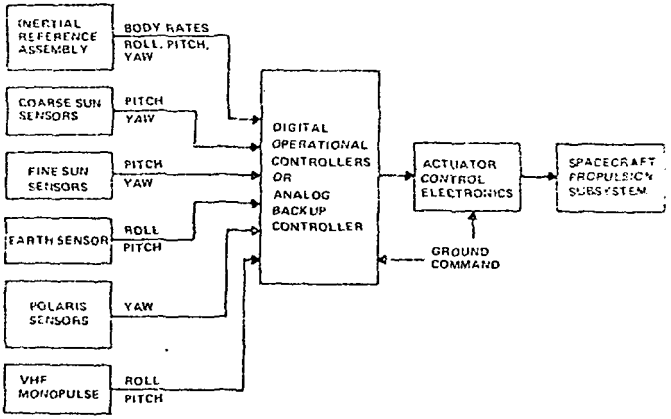


Figure II-11A. ACS Block Diagram (Acquisition Control Mode)

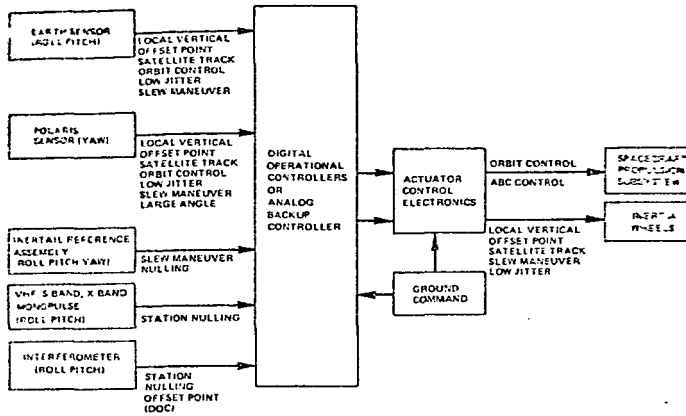

EARTH SENSOR (ROLL PITCH)
LOCAL VERTICAL OFFSET POINT SATELLITE TRACK ORBIT CONTROL LOW JITTER SLEW MANEUVER
POLARIS SENSOR (YAW)
LOCAL VERTICAL OFFSET POINT SATELLITE TRACK ORBIT CONTROL LOW JITTER SLEW MANEUVER LARGE ANGLE
INERTAIL REFERENCE ASSEMBLY ROLL PITCH YAW)
SLEW MANEUVER NULLING
VHF S BAND, X BAND MONOPULSE (ROLL PITCH)
STATION NULLING
INTERFEROMETER (ROLL PITCH)
STATION NULLING OFFSET POINT (DOC)
DIGITAL OPERATIONAL CONTROLLERS OR ANALOG BACKUP CONTROLLER
ACTUATOR CONTROL ELECTRONICS
ORBIT CONTROL
ABC CONTROL
LOCAL VERTICAL OFFSET POINT SATELLITE TRACK SLEW MANEUVER LOW JITTER
GROUND COMMAND

Figure II-11B. ACS Block Diagram (Operation Control Mode)

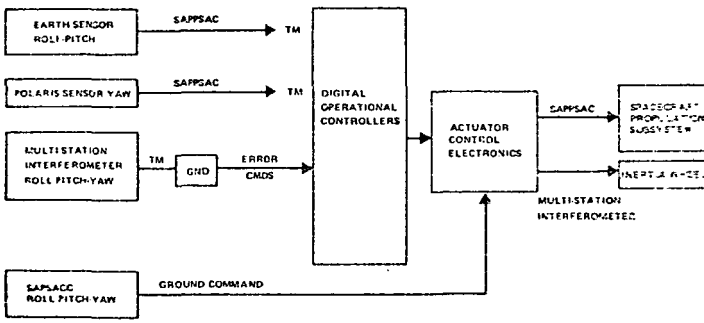


Figure II-11C. ACS Block Diagram (Experiment Mode)

In addition to performing control loop computations for stabilizing the spacecraft, the DOC provides the following functions:

- Accepts a wide variety of sensor inputs, solves control laws and commands torquers for all modes
- Executes acquisition logic to acquire the sun, earth, and Polaris
- Accepts commands for offset pointing (angle and ground coordinate commands)
- Computes compensations required for pointing commands to account for orbit eccentricity, orbit inclination, and Polaris diurnal motion
- Computes commands for performing Z-axis tracking of a low altitude satellite based on transmitted ATS and satellite ephemeris data
- Computes commands for performing slew maneuvers for antenna pattern measurements
- Provides a survival control mode if marooned in transfer orbit
- Provides a self-test routine within the DOC.

The sensor signals interface with the DOC (primary controller) as shown in Figures II-10 and II-11. The DOC drives the inertia wheels which are the primary precision torquer elements.

The basic DOC consists of a power supply, central processor, a memory, and an input/output section. The DOC provides for block transfer of data to the telemetry DACU or complete data transfer from memory on command from the ground.

The heart of the DOC is the central processor that performs programs stored within the memory. The input/output electronics accepts both analog and digital signals and can supply monitoring signals to telemetry, and control signals to sensors and to the ACE for controlling the wheels and jets.

Analog Backup Controller—The ABC is a simple, low power analog controller that serves as a backup to the DOC's in certain modes at reduced power levels. It consists of a power supply, mode logic, and switching amplifiers. A block diagram of the ABC is shown in Figure II-12. It provides the following functions:

- Stabilizes the spacecraft
- Acquires the sun, earth, and Polaris with aid from the ground
- Maintains local vertical hold mode (0.5 degree accuracy)
- Maintains a station pointing mode (±0.05 degree accuracy with C-band monopulse).

The ABC has no digital interfaces. The basic control loop consists of switching amplifiers with pseudo rate feedback compensation (when not using gyros) for stabilizing the loop. The SPS jets or wheels are used for attitude torquing. The ABC is selected by command from the ground and interfaces with the ACE jets and wheels.

The ABC uses analog error signals from the following combinations of sensors:

- Earth sensor analog outputs (roll and pitch)
- Polaris sensor (yaw, both units)
- IRA (analog rate outputs for rate damping)
- Coarse sun sensor and fine sun sensor (pitch and yaw)
- C-band, S-band, and VHF monopulse (roll and pitch)

Operational backup capability is shown in Table II-6, and a summary of ACS power characteristics is given in Table II-7. Figure II-13 shows the location of the components of the ACS.

Earth Sensor Assembly—The earth sensor is used for earth acquisition during the acquisition modes and serves as the primary roll/pitch sensor for all operational modes. The earth sensor assembly consists of two sensing heads (one roll, one pitch) plus an electronics unit. Each head contains a sensing element, electronics, scan mirror, and an offset mirror. The scan mirror is driven back and forth to provide a scan amplitude of ±13 degrees. An encoder is mounted on the mirror to indicate center crossing and provide pulses to a counter. Earth radiance signals gate the pulses to the counter. The scan mirror is biased by the cross-axis-attitude-angle signals to keep

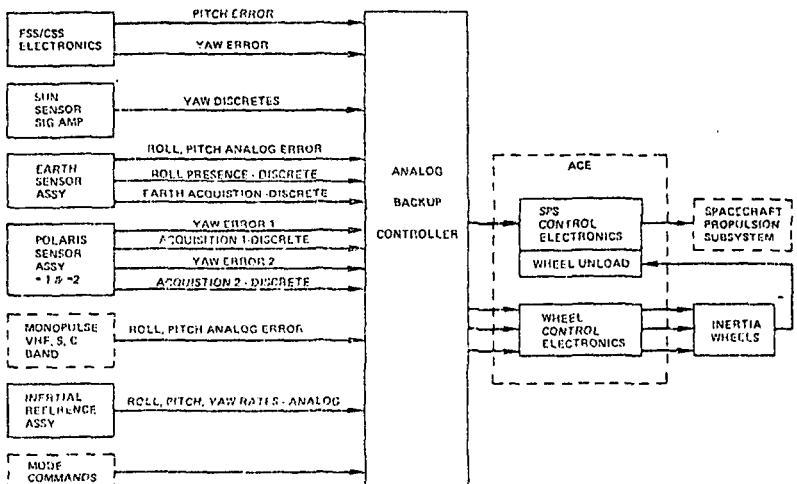


Figure II 12. Analog Backup Mode Block Diagram

Table II-6
Backup Capability—Operational

Sensors		Controllers	Torquers
R, H, Pitch	Yaw		
Earth sensor	Polaris sensor 1 or 2	Digital operational controller 1 or 2	Wheels-SPS 1 or 2
Interferometer	Polaris sensor 2 or 1	Digital operational controller 2 or 1	Wheels-SPS 2 or 1
Monopulse (C,S,VHF)	Yaw Gyro		SPS 1 or SPS 2
Earth sensor	Polaris sensor 1	Analog backup controller	SPS 1
Monopulse (C,S, VHF)	Polaris sensor 2		SPS 2
Earth sensor	Polaris sensor 1 or 2	Ground computation and command	Wheels-SPS 1
Interferometer	Polaris sensor 2 or 1		Wheels-SPS 2
Monopulse (C,S,VHF)	Interferometer		SPS 1
			SPS 2

Table II-7
ACS Characteristics Summary

		Launch/Ascent (6.6 hr.)			ACQUISITION MODES: Rate Damp/Coarse Sun Acquisition (10% Duty Cycle, 60 min.)			ACQUISITION MODES: Fine Sun/Earth/Polaris (10% Jet Duty Cycle, 150 min.)			OPERATIONAL MODES: Normal Orientation (Wheels & Jets)			OPERATIONAL MODES: Offset Point (Wheels & Jets)			OPERATIONAL MODES: Normal Orientation (Jets 0.1% Duty Cycle)			OPERATIONAL MODES: Orbit Control (2% Duty Cycle)					
		Peak	Norm.	Δ	Peak	Norm.	Δ	Peak	Norm.	Δ	Peak	Norm.	Δ	Peak	Norm.	Δ	Peak	Norm.	Δ	PEAK		NORMAL		Δ	
																				E-W	N-S	E-W	N-S	E-W	N-S
SPS Heaters, Transducers	D	20.8	20.8	0	20.8	10.8	0	20.8	10.8	0	20.8	10.8	0	20.8	10.8	0	20.8	10.8	0	20.8	20.8	10.8	20.8	0	0
SPS Valves AC	D	0	0	0	9	1	0	9	1	0	9	0	0	9	0	0	9	0	0	9	9	0	0	0	0
SPS Valves DC	D	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	3	6	3	6	0	0
SPS Total	D	20.8	20.8	0	29.8	11.8	0	29.8	11.8	0	29.8	10.8	0	29.8	10.8	0	20.8	10.8	0	32.8	35.8	13.8	26.8	0	0
Inertia Wheels	D	0	0	0	0	0	0	0	0	0	46.5	17.1	0	46.5	17.1	0	0	0	0	0	0	0	0	0	0
ACE	D	22.0	22.0	0	23.8	11.5	0	23.8	11.5	0	49.7	29.6	0	49.7	29.6	0	23.8	11.3	0	24.4	25.0	11.9	23.2	0	0
ACE[1] Total	I	42.8	42.8	0	53.6	23.3	0	53.6	23.3	0	126.0	57.5	0	126.0	57.5	0	53.6	22.1	0	57.2	60.8	25.7	50.0	0	0
Sun Sensors & Electronics		0	0	0	3.6	3.6	0	3.6	3.6	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Earth Sensor & Electronics[3]		0	0	0	0	0	0	23.3	20.2	0	15.6	15.6	0	23.3	20.2	0	15.6	15.6	0	15.6	15.6	15.6	15.6	0	0
Polaris Sensors & Sun Shutters[2]	D	13.0	0	0	13.0	0	0	20.0	7.0	0	20.0	7.0	0	20.0	7.0	0	20.0	7.0	0	20.0	20.0	7.0	7.0	0	0
3-Axis IRA[2]	D	0	0	0	40.0	40.0	0	40.0	40.0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
ABC/SCE	D	1.3	0.6	0	8.3	7.6	0	12.0	11.3	0	5.0	4.3	0	5.0	4.3	0	5.0	4.3	0	5.0	5.0	4.3	4.3	0	0
ABC/SCE	I	14.3	0.6	0	61.3	47.6	0	72.0	58.3	0	25.0	11.3	0	25.0	11.3	0	25.0	11.3	0	25.0	25.0	11.3	11.3	0	0
DOC		0	0	0	26.9	26.9	0	26.9	26.9	0	26.9	26.9	0	26.9	26.9	0	26.9	26.9	0	26.9	26.9	26.9	26.9	0	0
ACS/SPS TOTAL[1]	I	57.1	43.4	0	145.4	101.4	0	179.4	132.3	0	193.5	111.3	0	201.2	115.9	0	121.1	75.9	0	124.7	128.3	79.5	103.6	0	0

Notes: (1) Each Power entry represents both input (I) and dissipated (D) power unless otherwise noted.

(2) When the IRA with sun sensor drift trim is used in lieu of the Polaris Sensor, the Polaris Sensor input/dissipated power becomes 0 watts, the IRA power becomes 47 watts, and the sun sensor power becomes 4.6 watts.

(3) When the interferometer is used in lieu of the Earth Sensor, the input/dissipated power of the Earth Sensor becomes 0 watts and the interferometer power becomes 15.5 watts; when the monopulse is used in lieu of the Earth Sensor, the Earth Sensor input/dissipated power becomes 0 watts.

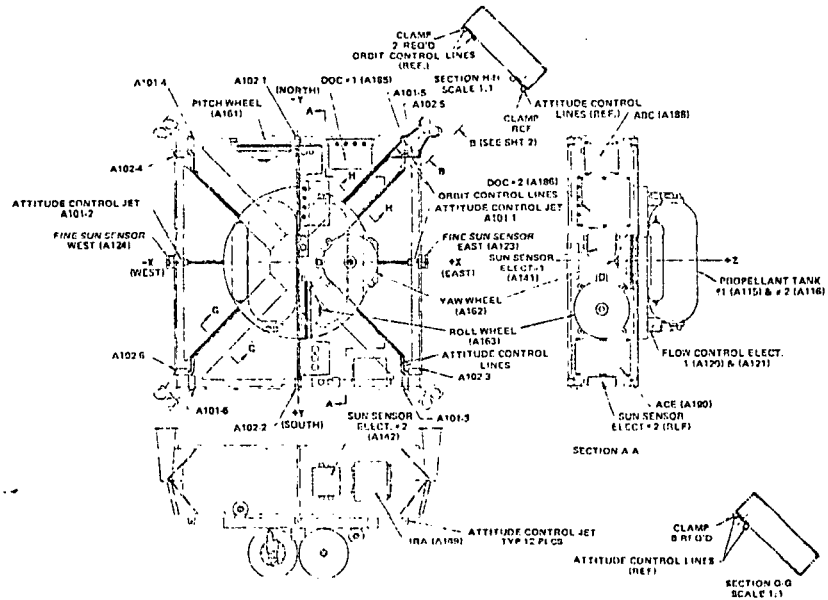


Figure II-13. Attitude Control System Component Location

the scan centered on the earth. The roll-offset mirror is biased by the pitch-attitude-angle signals to keep the roll scan near the full diameter of the earth. The pitch-offset mirror is biased by the roll-attitude-angle signal to keep the pitch scan near the full diameter.

The earth-sensor-attitude-angle output is formed by scanning the instantaneous field-of-view of the bolometer telescope assembly across the earth disk. During the scan, encoder pulses representing incremental angle changes are gated into an up-down counter in the earth sensor electronics based on the following events:

- Space-to-earth crossing
- Center crossing on the encoder (Z-axis null plane)
- Earth-to-space crossing

The attitude output is determined after the scan has completed a full cycle. Thus, with a scan frequency of 4 Hz, new outputs are available every 0.25 second. This output is available as a digital output. An analog signal is also available as a result of the signals generated to bias the scan and offset mirrors.

The attitude output of the earth sensor is supplied to the two DOC's (digital) and the ABC (analog), as well as the two DACU's (digital) for telemetering to the ground. An automatic sun avoidance mode is also provided in the sensor.

Polaris Sensor Assembly—Two Polaris sensors and their respective conditioning electronics are provided. The ground will choose which sensor is to be used. The Polaris sensor is used during the Polaris acquisition mode and serves as the primary yaw sensor for all operational and most experimental modes. The Polaris sensor responds to irradiation from Polaris and provides an analog output voltage proportional to the angle between the line-of-sight to Polaris and the place containing the Y-Z axes of the sensor. The sensor has a linear range of ±5.5 degrees in the sensitive axis. The field-of-view in the nonsensitive direction is greater than ±1.5 degrees. To make use of the full tracking field, the sensor has the capability, on command, of electronic gimbaling (in roll) of the instantaneous field-of-view retaining the star in the field-of-view during roll maneuvers.

Inertial Reference Assembly—The three-axis inertial reference assembly (IRA) is used during the acquisition mode and the acquisition of Polaris. It can

provide some temporary attitude hold functions and is the primary 3-axis sensor for the transfer orbit failure mode. The IRA consists of three gas-bearing spinmotor gyros, with associated electronics, that provide output pulses corresponding to incremental body angle changes about each of the three axes. By counting the pulses, the total attitude can be calculated. The DOC also uses these pulses in its computations to provide body rate signals for rate damping and acquisition modes. Filtering of the signals is accomplished in the ABC.

The electronics for the IRA include:

- A power supply
- Four temperature control amplifiers, one for the oven and three for the gyros
- Three pulse-rebalance loops

The IRA provides an analog rate output to the ABC and the two DACU's. The attitude outputs will be transmitted to ground as a part of the data block transfer from the DOC.

Coarse-Null Sun Sensors—Four coarse-null sun sensors are provided, each with two-axis sensitivity. Each sensor makes use of silicon photovoltaic diode eyes. Each eye provides an analog output signal related to the angle between the sun line and a vector normally incident on the sensor. When used in conjunction with the electronics, the output from two differentially connected eyes will provide a null when the sun is at equal angles relative to each of them.

The functions of the coarse-null sun sensors are to enable the spacecraft X-axis to be brought to within 8 degrees of the sun line, where the fine sun sensor can take over control of the spacecraft.

The coarse sun sensors are located on the hub and the earth viewing module (EVM), providing a 4π steradian field-of-view. The yaw coarse sun sensor experiences some blockage by the solar panels, requiring two additional coarse eyes on the hub to provide for sun line location capability throughout the yaw plane.

The coarse sun sensors, through their electronics, interface with the DOC and ABC for onboard use and with the DACU's for telemetry to the ground.

Fine-Null Sun Sensor—Two fine-null sun sensors are provided, each with two-axis sensitivity. Each sensor consists of four silicon photovoltaic diode eyes interconnected to obtain a null output when the sensors are pointing to the sun. In addition, a target eye with a field view of approximately ±8 degrees is included to provide an indication to the controllers for switching control from the coarse sensors to the fine sensors. The fine sensors have an accuracy of ±2 minutes of arc.

Monopulse—The antenna system operates in a monopulse mode at C-band. In this mode a pointing error signal is generated for the roll-yaw and pitch-yaw planes with an angular accuracy of 1/50 beamwidth. This signal interfaces with the attitude control subsystem and the telemetry subsystem. The monopulse also operates at S-band for closed-loop tracking of DRS satellites and at VHF for initial wide-angle ground station acquisition.

Actuator Control Electronics—The actuator electronics (ACE), which is energized at all times for all modes of operation, includes the wheel drive electronics, the wheel unload logic, the SPS control electronics and associated power supplies, and the signal conditioning circuitry. The wheel drive electronics drives the inertia wheels in response to attitude error signals from the DOC's, ABC, or one of the two CDD's or two GAC decoders.

The ACE serves as the power amplifier between the controllers and the inertia wheels or the SPS. The ac wheels are driven by means of pulse modulation. A pulse modulator as exists in the ACE accepts analog signals as generated by the DOC or the ABC and gates the wheel full on or off at a duty cycle which is proportional to spacecraft error. Outputs (2¢400∼square wave) are supplied to both phases of the motor.

The electronics for the SPS consist of the logic to accept the jet firing commands from the two DOC's, the ABC, the two command decode and distribution (CDD) units, and the power drivers to energize the appropriate solenoids in the SPS for attitude control. The electronics also provide the power required for switching the latching valves and for the heaters in the SPS. Additional electronics are required to supply heater and solenoid power for the orbit control thrusters in the SPS.

The ACE provides outputs to 12 attitude control thrusters and 4 orbit control thrusters. The wheel drive section has outputs to three inertia wheels. The ACE also processes the wheel tachometer signals required for control compensation, for wheel unloading, and for telemetry to ground.

SPS Control Electronics—The SPS control electronics supplies the necessary signals for control of the:

- solenoid valves for the attitude control and orbit control thrusters
- latching valves
- transducers, and
- heaters for the tanks, lines, valves, and catalyst beds.

The SPS control electronics supplies commands to the

- attitude control thrusters, and
- orbit control thrusters.

Ground control commands to the ACE determine the thrusters to be commanded ON for the required attitude control and orbit control action.

The block diagrams in Figures II-14 and II-15 illustrate the SPS and the interface signals. Table II-8 lists the input interface signals from the ACE to the SPS.

Inertia Wheels—Three inertia wheels serve as the prime torquers for all modes of operation except acquisition, orbit control and, jet only control. These wheels form a part of the ACS (one roll, one pitch, and one yaw). A pulse-type tachometer is provided for use by the DOC to obtain spacecraft rate compensation for the control loop. The tachometer output is also used to indicate the need for inertia wheel unloading.

The inertia wheels have an ac, low-resistance squirrel cage motor driven by a pulse modulator. The drive electronics provide a variable duty cycle of 400 drive voltage which is a function of spacecraft error. This provides a command acceleration, resulting in a torque that reacts against the spacecraft to rotate it.

Each wheel is designed to provide 0.03 ft-lb of torque and 10-lb/sec of momentum storage at 1500 rpm.

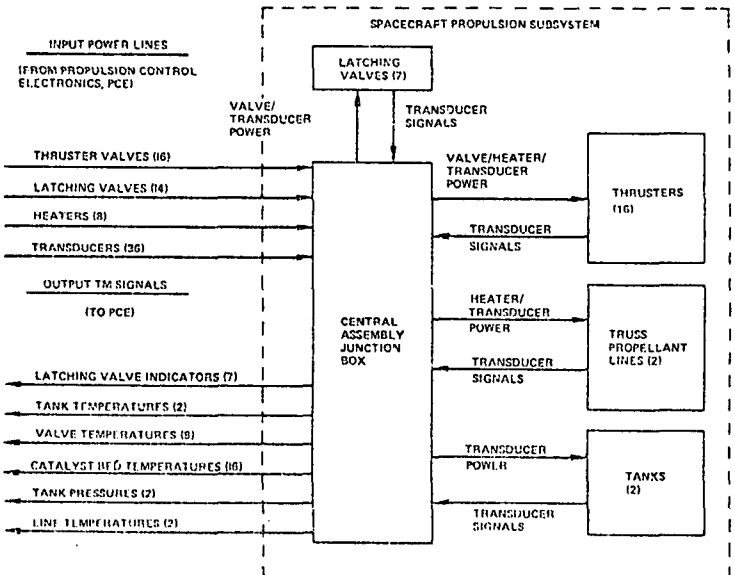


Figure II-14. Interface Signals - Spacecraft Propulsion Subsystem

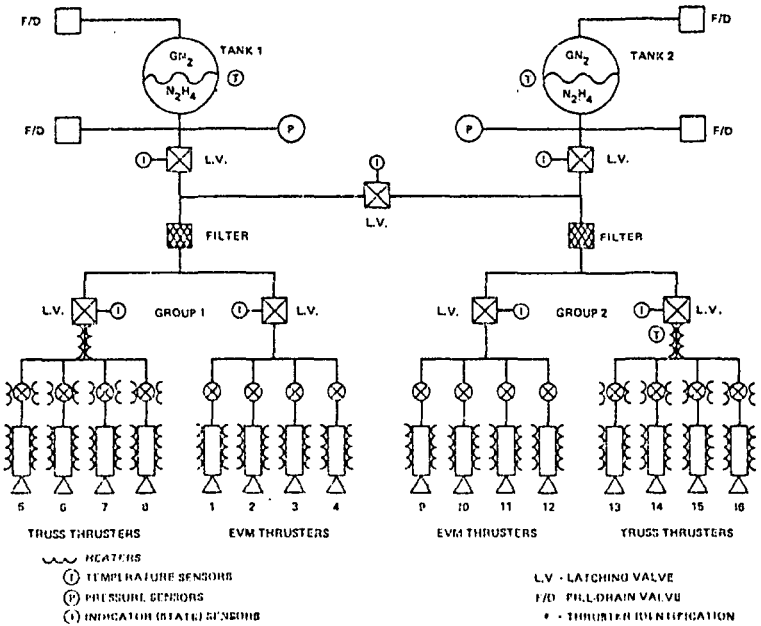


Figure II-15. Spacecraft Propulsion Subsystem Diagram

Table II-8

Input Interface Signals - SPS From ACE

Item	Signal Description	Item	Signal Description
1	Thruster Valve ON-OFF, Thruster No. 1 (E-Y-1)	16	Thruster Valve ON-OFF, Thruster No. 16 (T-X-X-2
2	Thruster Valve ON-OFF, Thruster No. 2 (E+Y-1)	17	Latching Valve Open, Tank #1
3	Thruster Valve ON-OFF, Thruster No. 3 (E+X-1)	18	Latching Valve Closed, Tank #1
4	Thruster Valve ON-OFF, Thruster No. 4 (E-X-1)	19	Latching Valve Open, Tank #2
5	Thruster Valve ON-OFF, Thruster No. 5 (E-Y-2)	20	Latching Valve Closed, Tank #2
6	Thruster Valve ON-OFF, Thruster No. 6 (E+Y-2)	21	Latching Valve Open, Truss Group #1
7	Thruster Valve ON-OFF, Thruster No. 7 (E+X-2)	22	Latching Valve Closed, Truss Group #1
8	Thruster Valve ON-OFF, Thruster No. 8 (E-X-2)	23	Latching Valve Open, Truss Group #2
9	Thruster Valve ON-OFF, Thruster No. 9 (T+X-Y-1)	24	Latching Valve Closed, Truss Group #2
10	Thruster Valve ON-OFF, Thruster No. 10 (T-X-Y-2)	25	Latching Valve Open, EVM Group #1
11	Thruster Valve ON-OFF, Thruster No. 11 (T+X+Y-1)	26	Latching Valve Closed, EVM Group #1
12	Thruster Valve ON-OFF, Thruster No. 12 (T-X+Y-2)	27	Latching Valve Open, EVM Group #2
13	Thruster Valve ON-OFF, Thruster No. 13 (T+X+X-1)	28	Latching Valve Closed, EVM Group #2
14	Thruster Valve ON-OFF, Thruster No. 14 (T-X+X-2)	29	Latching Valve Open, Cross Strap
15	Thruster Valve ON-OFF, Thruster No. 15 (T+X-X-1)	30	Latching Valve Closed, Cross Strap

Table II-8

Input Interface Signals - SPS From ACE (Cont)

Item	Signal Description	Item	Signal Description
31	Heaters for truss lines and valves, Group #1 (Power)	45	Thermocouple for Thruster No. 7
32	Back-up Heaters for truss lines and valves, Group #1 (Power)	46	Thermocouple for Thruster No. 8
33	Heaters for truss lines and valves, Group #2 (Power)	47	Thermocouple for Thruster No. 9
34	Back-up Heaters for truss lines and valves, Group #2 (Power)	48	Thermocouple for Thruster No. 10
35	Heaters for 8 thruster catalyst beds, Group #1 (Power)	49	Thermocouple for Thruster No. 11
36	Heaters for 8 thruster catalyst beds, Group #2 (Power)	50	Thermocouple for Thruster No. 12
37	Back-up Heaters for 8 thruster catalyst beds, Group #1 (Power)	51	Thermocouple for Thruster No. 13
38	Back-up Heaters for 8 thruster catalyst beds, Group #2 (Power)	52	Thermocouple for Thruster No. 14
39	Thermocouple for Thruster No. 1	53	Thermocouple for Thruster No. 15
40	Thermocouple for Thruster No. 2	54	Thermocouple for Thruster No. 16
41	Thermocouple for Thruster No. 3	55	Thermistor for Truss Line 1
42	Thermocouple for Thruster No. 4	56	Thermistor for Truss Line 2
43	Thermocouple for Thruster No. 5	57	Thermistor for Tank Line 1
44	Thermocouple for Thruster No. 6	58	Thermistor for Tank Line 2
		59	Thermistor for Thruster Valve 5
		60	Thermistor for Thruster Valve 6
		61	Thermistor for Thruster Valve 7

Table II-8

Input Interface Signals - SPS From ACE (Cont)

Item	Signal Description	Item	Signal Description
62	Thermistor for Thruster Valve 8	68	Pressure Transducer, Tank No. 2 (Power)
63	Thermistor for Thruster Valve 13	69	Latching Valve Indicator
64	Thermistor for Thruster Valve 14	70	Latching Valve Indicator
65	Thermistor for Thruster Valve 15	71	Latching Valve Indicator
66	Thermistor for Thruster Valve 16	72	Latching Valve Indicator
67	Pressure Transducer, Tank No. 1 (Power)	73	Latching Valve Indicator
		74	Latching Valve Indicator
		75	Latching Valve Indicator

SPACECRAFT PROPULSION SUBSYSTEM

The Spacecraft Propulsion Subsystem (SPS) provides thrusts of suitable magnitude and direction to torque the spacecraft about the three axes for attitude control (Figure II-16). The SPS also provides thrusts to impart changes in orbital velocity of the spacecraft for east-west and north-south orbit control.

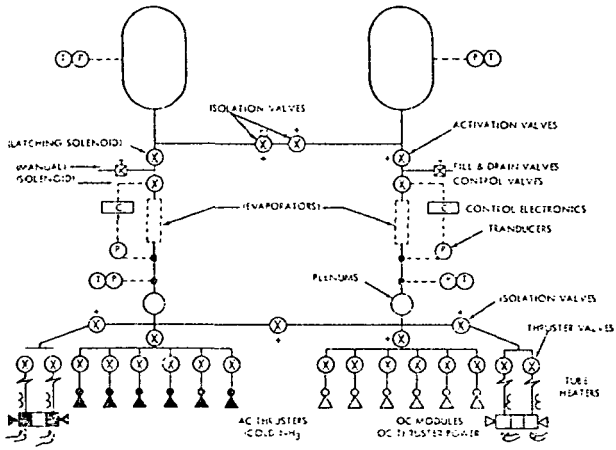


Figure II-16. *Selected Spacecraft Propulsion Subsystem*

The SPS uses hydrazine for attitude and orbit control. It consists of two (standby redundant) regulated pressure feed assemblies, 12 unheated attitude control thrusters, and two resistively heated orbit control thruster modules, all controlled by electronics in the actuator control electronics.

The attitude control resistojets are mounted on the EVM. The orbit control jets are mounted on the reflector support truss in the plane of the spacecraft center of gravity (Figure II-17). A dual-nozzle thruster is located

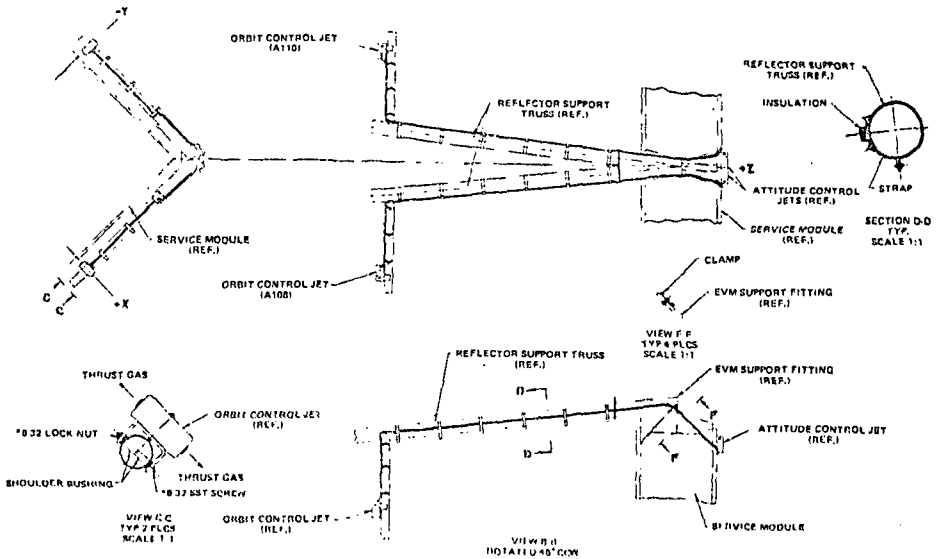


Figure II-17. Orbit Control Thrusts Locations

on the +X axis for east-west thrusting and on the +Y axis for north-south thrusting. Strip heaters are provided on the propellant lines to prevent freezup.

The propellant is stored in two cylindrical tanks with a latching valve in the tie line between them to provide isolation in the event of a leak.

Table II-9 tabulates the SPS mission requirements and the layout of the SPS within the EVM is shown in Figure II-18.

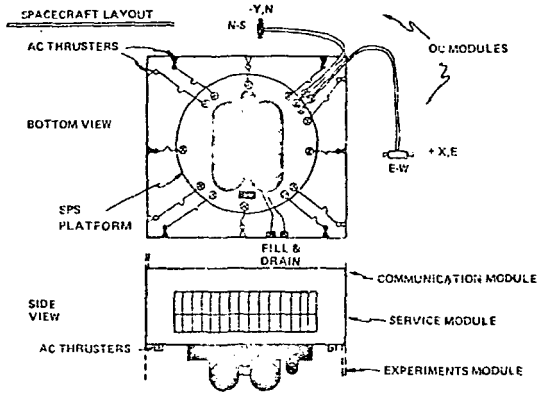


Figure II-18. SPS Layout within EVM

Table II-9
SPS Mission Requirements

Item	Characteristic Velocity (ft/sec)	Impluse (Lb-Sec)		Thrust Time
		Design S/C (2000 lbs.)	Max Wt S/C (2450 lbs)	(Design S/C. T = 0.003 lb-
Initial Orbit Adjust	30	1875	2295	7.2 days
East-West Station Keeping (2 years)	14	870	1070	3.5 hrs. (ea. month)
North-South Station Keeping (1 month)	14	870	1070	1.35 hrs. (ea. node every 12 hrs.)
East-West Station Repositioning (Transfer from 94° West to 15° East - 41.2 days)	60	3750	4570	7.2 days (each end)
Additional Orbit Maneuvers	10	635	765	---
TOTAL	128	~8000	~9800	

Item	Design S/C	Max. Wt. S/C
Orbit Control Propellant Wt (Lbs.) (I_{SP} = 200 sec.)	40	49
Attitude Control Propellant Wt (lbs) (I_{SP} = 100 sec.)	25	29
Total Propellant Wt. (Lbs.)	66	78

ELECTRICAL POWER SUBSYSTEM

Functional Operation

The Electrical Power Subsystem derives power from two hemi-cylindrical solar array panel assemblies that are located on the Y axis of the spacecraft and extend beyond the 30-foot reflector to avoid shadowing. Figure II-19 is a simplified block diagram of this subsystem.

Energy storage is provided by two 19-cell, 15 A-H, nickel-cadmium batteries which are discharged in parallel. The batteries provide electrical power during occult portions of the ascent phase, and peak loading that exceeds the solar array capability.

The power output from the solar array and batteries is regulated to a voltage level of 30.5 vdc for use by the spacecraft loads. The electrical power system uses partial shunt regulation for regulating the solar array voltage and a boost regulator for regulating the battery voltage.

When excess power is available from the solar array, the battery chargers are enabled to recharge the batteries. Each battery has separate charge control circuits and are normally charged with a current limited (1.5 amp maximum, C/10) rate with temperature compensated voltage control. When the array power exceeds the load power and battery charge requirements, the excess power is dissipated by shunt circuits.

A common controller provides the necessary error signals for the three major modes of operation; the shunt mode, the charge mode, and the boost mode. These functional modes provide the regulation and charge control for the electrical power subsystem. Figure II-20 shows these control modes.

Secondary power conditioning at 28 vdc is provided in the electrical power subsystem for the rf solid state power amplifiers and for each GFE experiment through individual load interface circuits.

Power Subsystem Capability/Requirements

The power subsystem is sized to provide continuous peak power of 525 Watts for a minimum of 4 hours (using batteries to a maximum of 50% depth of discharge (DOD), 170 Watt hours per battery) and a minimum average output of 450 Watts for the remaining 20 hour operational day through 2 years in orbit except during occult. A minimum of 290 load Watt hours (50% DOD) is available from both batteries for operation during occult.

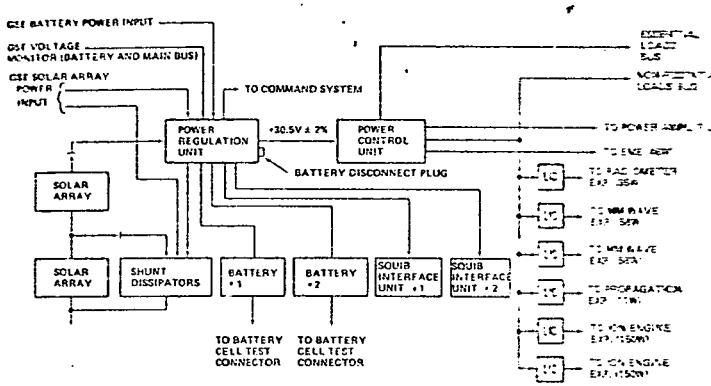


Figure II-19. Power Subsystem Simplified Block Diagram

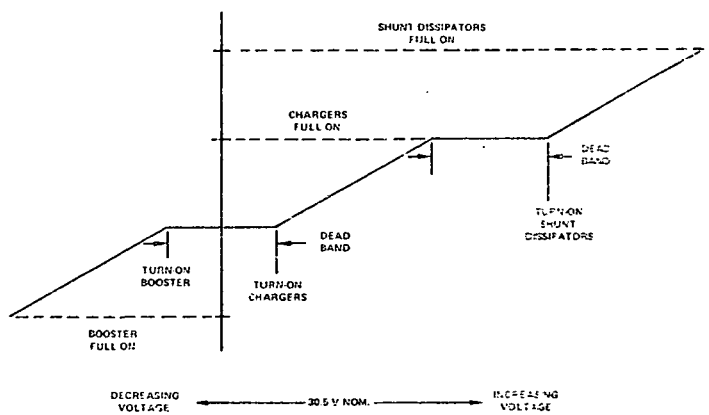


Figure II-20. Power Subsystem Regulation Control Modes

The power subsystem capability is shown below:

- Battery power available to loads: 290 Watt hours (50% DOD/battery)
- Solar Array power (Watts):

Years-	0	1	2	5
a) Equinox	645	550	510	450
b) Summer solstice +4° offset pointing	560	490	460	415

- Solar Array/Battery share mode at end of 2 yrs (Watts):

Hours-	2	3	4
a) Equinox	655	610	585
b) Summer Solstice +4° offset pointing	605	560	535

Power is provided for the spacecraft subsystems at the following bus voltages:

a) 30.5 ±2% at the power control unit output for use by spacecraft subsystems.

b) 28 ±2% at the power control unit output for use by the communication subsystem.

c) 28 ±2% at the experiment inputs except for EME, which is 28 ±8%.

d) 30 ±5% and 19 ±2V at the power regulation unit output for use by the command decoders and receivers respectively.

Power Subsystem Protective Features

To protect the power subsystem and loads from anomolies that may occur, the following protection features are incorporated in the power subsystem:

ANOMALY	RESPONSE
• Nonessential bus under voltage/overload	Disconnect nonessential loads
• Power amplifier bus overload	Current limit with overload disconnect
• Experiment bus overload	Current limit with overload disconnect
• Battery under voltage	Disconnect battery
• Battery over-temperature	Disconnect charger
• PRU bus high/low voltage	Switch to redundant functions
• PRU booster overload	Disable booster
• PRU bus high voltage (38 V)	Over-voltage clamp circuit limits voltage

Power Subsystem Components

The power subsystem consists of the following components:

- Solar Array Panels
- Power Regulation Unit
- Power Control Unit
- 19-Cell Ni-Cd Batteries
- Shunt Dissipators
- Load Interface Circuits (LIC)
- Squib Interface Units

Figure II-21 is a detailed block diagram of the Power Subsystem showing the circuit functions with redundancy within the hardware units.

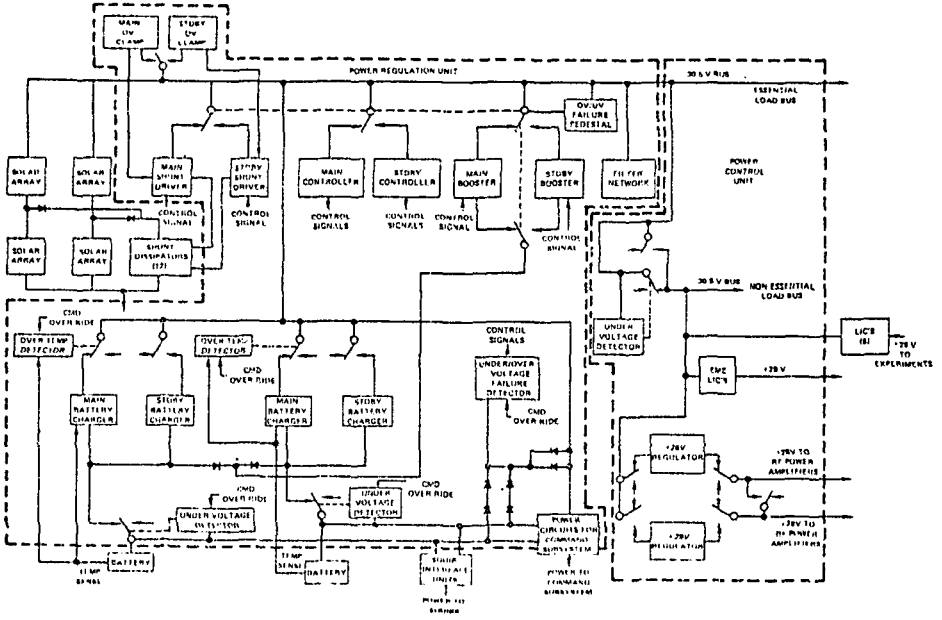


Figure II-21. Power Subsystem Functional Block Diagram

Solar Array—Pertinent facts on the mechanical configuration of the array follow:

Number of cells	21,600
Number of parallel cells per string	3 (2 cm x 4 cm)
Number of series cells per string	75
Number of strings	96
Number of panels	32
Mechanical area	216 sq ft
Solar cell area	186 sq ft
Packaging efficiency	86%

The solar array output power is a nominal 600 watts initially decreasing to a nominal 485 watts after 2 years.

Each solar array panel contains solar cells (675) which have the following characteristics:

1) N/P silicon

2) 2 x 4 cm

3) 2 ohm-cm nominal resistivity

4) 11.0% minimum average efficiency

5) 12 mil thickness

The solar array design is based on a radiation degradation factor of 20% for a two year mission with a 6-mil radiation protection cover glass of microsheet.

Power Regulation Unit—The power regulation unit provides the following major functions for the power subsystem:

1) Main bus regulation at 30.5 Volts ±2%

2) Overload, overvoltage, and undervoltage automatic protection on the main bus

3) An undervoltage sensing and disconnect function for each battery

4) Redundancy with the following circuit functions:

   a) Battery charge regulators

   b) Battery discharge boost regulator (booster)

   c) Shunt driver

   d) Controller

5) Operating mode controls including on/off control for the battery charges

6) A battery connect/disconnect function for each battery

7) Voltage, current, temperature, and status monitors for telemetry

8) An input for external power during the prelaunch phase for solar array and battery simulation

9) Command and automatic control of redundant switching and failure mode switching

10) A battery charger disconnect to remove the battery from charge when the battery temperature reaches 35° C ± 2° C.

11) A disconnect plug to arm and disarm the batteries from the power subsystem.

Power Control Unit—The power control unit provides the following functions for the power subsystem:

1) A +28 v +2%, -3% for the EME with a load interface circuit within the power control unit

2) An undervoltage circuit breaker for the non-essential loads

3) A distribution point for routing the regulated voltage to the essential and non-essential loads

4) Redundant of the +28 volt regulators with ±2% regulation and over-current protection for the power amplifiers

5) Voltage, temperature, and status monitors for telemetry

Batteries—Two 15 ampere-hour nickel-cadmium batteries are provided to supply electrical power during occult, portions of the ascent phase, and peak loading exceeding the solar array capability.

The batteries are sized to provide sufficient power for the occult and load sharing modes without exceeding a 50% DOD (170WH/battery) from rated capacity on a daily basis when operated in the normal two-battery mode.

Some additional solvent battery characteristics are the following:

Nominal stored energy : 342 Watt hrs per battery

Nominal discharge voltage : 22.8 Volts

Maximum charge voltage : 28.5 Volts

Minimum discharge voltage : 19.0 Volts

Shunt Dissipators—The shunt dissipators are designed for a maximum dissipation of 50 watts and a continuous nominal dissipation of 35 watts per unit. The shunt dissipators perform their functions of regulating by drawing current from a tap section of the solar array. The dissipators receive a linear control signal proportional to the regulated bus voltage error from the shunt driver located in the power regulation unit.

Load Interface Circuits—The load interface circuits shall be designed to handle three load ranges. (See Figure II-22.) These load ranges are:

1) 0 to 15 watts

2) 0 to 60 watts

3) 0 to 150 watts

These load interface circuits control the interface between the 30.5-volt bus and the GFE experiment loads, and provide the following functions:

1) +28.0 volts ±2% output except for the EME, which is +28.0 volts +2%, -3% output

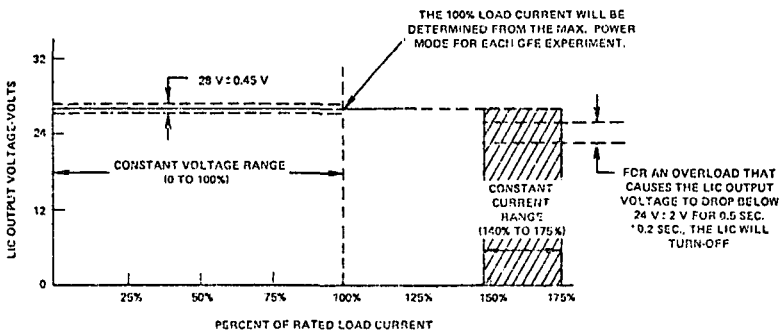

THE 100% LOAD CURRENT WILL BE DETERMINED FROM THE MAX. POWER MODE FOR EACH GFE EXPERIMENT.
28 V ± 0.45 V
CONSTANT VOLTAGE RANGE (0 TO 100%)
CONSTANT CURRENT RANGE (140% TO 175%)
FOR AN OVERLOAD THAT CAUSES THE LIC OUTPUT VOLTAGE TO DROP BELOW 24 V ± 2 V FOR 0.5 SEC. ± 0.2 SEC., THE LIC WILL TURN-OFF
LIC OUTPUT VOLTAGE-VOLTS
32
24
12
0
25%
50%
75%
100%
125%
150%
175%
PERCENT OF RATED LOAD CURRENT

Figure II-22. LIC Performance Characteristics

2) Line isolation

3) Overload protection

4) ON/OFF command control

5) Undervoltage protection

The load interface circuit for the EME is contained within the power control unit.

Squib Interface Units—The squib interface unit provides for the safing and arming of electroexplosive devices, for the actuation of these devices, and for monitoring by telemetry. The units also provide the capability for interrogating pyrotechnic and activator circuits for validation purposes. Current limiting is provided for each squib circuit. A simplified block diagram of the pyrotechnic system is shown in Figure II-23.

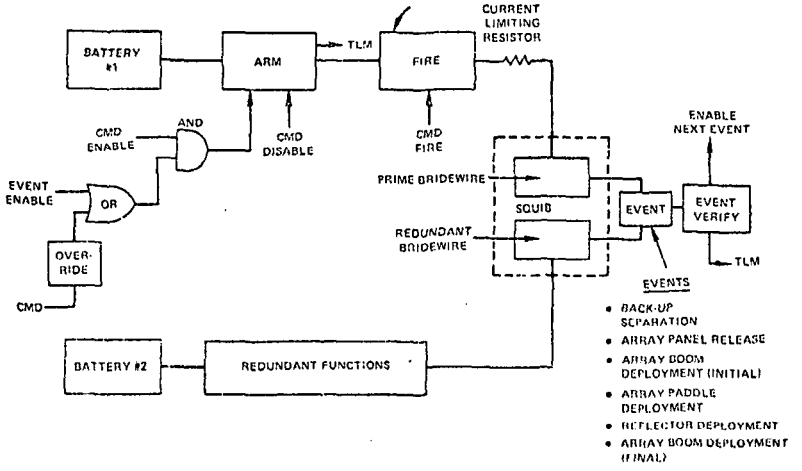


Figure II-23. Pyrotechnic Functional Diagram

COMMUNICATIONS SUBSYSTEM

The operation of the Communications Subsystem is essential to the successful performance of all the experiments on the ATS spacecraft and to the accomplishment of the primary objective of the ATS-F and -G mission. The transponder provides the basic interface between the experiments in the satellite and ground terminals. In some instances the transponder is itself a part of the experiment. Figure II-24 is a block diagram of the subsystem. Table II-10 describes the subsystem characteristics. The subsystem layout is shown in Figure II-25.

The Communications Subsystem operates to fulfill the requirements of the following modes of operation and experiments:

Coherent Mode
Noncoherent Mode
PLACE Experiment
Data Relay Experiment
ITV Experiment
Millimeter Wave and Laser Reception
VHR Radiometer Data Transmission
Radio Beacon
Monopulse
RFI Experiment

In the coherent mode all local oscillator signals are derived from a single oscillator phase-locked to the C-band signal carrier transmitted from the ground to the spacecraft.

In the noncoherent mode, the local oscillator frequencies are generated within the spacecraft by a highly stable fixed-frequency oscillator with an initial frequency tolerance of ± 10 PPM and a long term stability of better than ± 3 PPM in three months.

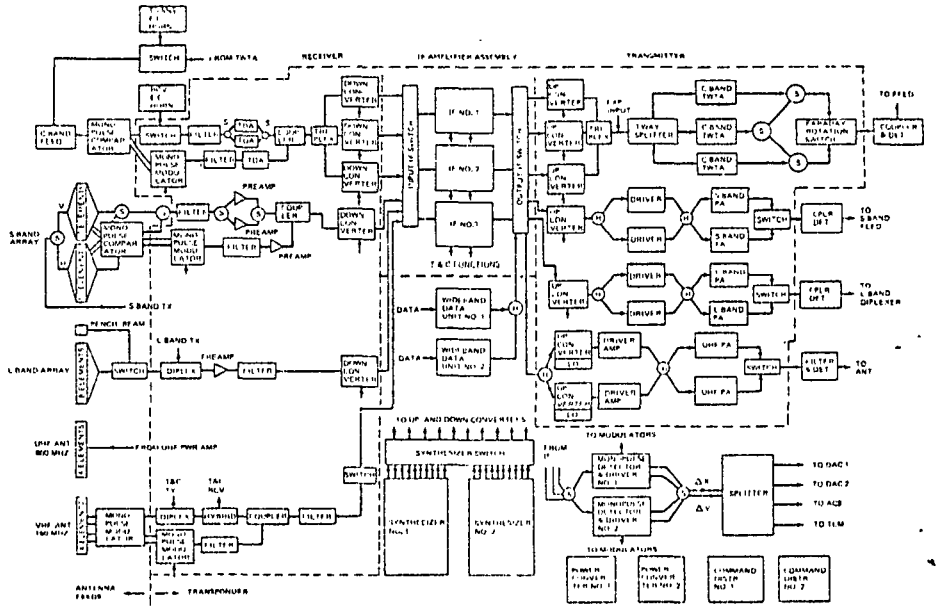


Figure II-24. Communication Subsystem Block Diagram

Table II-10
Communications Subsystem Characteristics

							RECEIVER		TRANSMITTER		
MODE	USER	NOMINAL FREQUENCY (MHz)	BANDWIDTH (MHz)	POLARIZATION	ANTENNA BEAMWIDTH (DEGREES)	PEAK ANTENNA GAIN (dB)	MIN G/T (DEGREES) (dB/K)	C/T (DEGREES) (dB/K)	TRANSMITTER OUTPUT POWER (WATTS)	MIN. EIRP (DEGREES) (dBW)	EIRP (PEAK) (dBW)
30 FOOT C-BAND RECEIVE	LASER MMW MONOPULSE	6350 5150 5950	40 12	LINEAR	0.4	49.0	30.5	13.5	-		
30 FOOT C-BAND TRANSMIT	LASER MMW ANTENNA TEST	3750 3950 4150	40	LINEAR	0.6	46.0	-	-	31.0	51.5 (1) 47.3 (2)	74.5 (1) 70.3 (2)
HORN C-BAND RECEIVE	TADRS ITV PLACE MMW LASERSATSR	6350 6150 5450	40 12	LINEAR	20	19.5	-20	-17	-	-	-
HORN C-BAND TRANSMIT	TADRS BEACON PLACE MMW LASER RADIOMETER ITV	3950 3750 4150 3450	40 500	LINEAR LINEAR	20 20	19.4 19.7	-	-	31.0 30.0	33.0 (1) 30.7 (2)	53.0 (1) 33.7 (2)
30 FOOT C-BAND RECEIVE	RSI	6150	500	HORIZONTAL VERTICAL RCP	0.4	48.5	NA	NA	NA	NA	NA
30 FOOT S-BAND RECEIVE SCAN	TADRS	2250	40	RCP	13.2 (3)	40.5	-	-	-	-	-
30 FOOT S-BAND TRANSMIT ON AXIS	TADRS	2490	12	RCP		39.5	-	-	20.0	-	59.5
30 FOOT S-BAND TRANSMIT SCAN	TADRS	2490	12	RCP	13.2 (3)	39.0	-	-	20.0	48	-
30 FOOT S-BAND RECEIVE ON AXIS	TADRS	2250	12 40	RCP		40.5	-	9.5	-	-	-
30 FOOT L-BAND PENCIL BEAM RECEIVE	PLACE	1650	12	RCP	3.5	38.5	2.5	5.5	-	-	-
30 FOOT L-BAND PENCIL BEAM TRANSMIT	PLACE	1530	12	RCP	3.5	38.5			40.0	48.0	91
30 FOOT L-BAND FAN BEAM RECEIVE	PLACE	1650	12	RCP	3.5 X 15	31.5	6.0	3			
30 FOOT L-BAND FAN BEAM TRANSMIT	PLACE	1530	12	RCP	3.5 X 15	31.5	-	-	40.0	48.0	91
30 FOOT SHF TRANSMIT	SSA	8450	500	RCP	1.0	53.0			800	48.0	91
30 FOOT SHF RECEIVE	MONOPULSE	7450	6	LINEAR	0.5	57	30	10			
30 FOOT SHF RECEIVE	TDRSMMW	17.8 to 18.1	83	LINEAR	0.5	57	40	10			
3 FOOT SHF TRANSMIT	TDRSMMW	17.8 to 18.1	5	LINEAR	0.5	57			1.0	37	99

For the PLACE, a special multiple access capability is provided for simultaneous C-band to L-band and L-band to C-band operation. The receiver linearity is such that received multiple carriers do not produce significant intermodulation components. The received multiple carriers are translated to baseband and then phase modulated onto a single carrier. Selectable baseband filters of 0.1 MHz and 1.0 MHz are provided. Automatic gain control maintains the phase modulation at a constant 1 radian independent of the number of carriers.

In performing the Data Relay Experiment, simultaneous S-band to C-band and C-band to S-band operation is provided. Also the C-band to S-band signal can be repeated on C-band. Emergency voice signals from the AAP satellite can be received and retransmitted.

Signals for the Instructional Television experiment are received at C-band and simultaneously transmitted on UHF and C-band. The UHF transmission is for the primary experimental use while the C-band signal can be monitored at ground stations geographically remote from the area covered by the UHF beam.

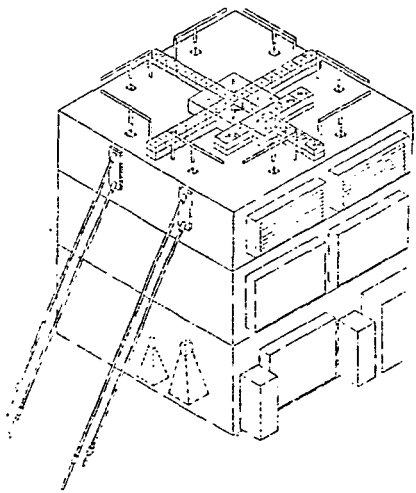


Figure II-25. EVM Layout Showing Communication Subsystem Location

For the wideband signals used in the Millimeter Wave and Laser experiments, the C-band receiver is used with a demodulator which is essentially flat from 30 Hz to 5 MHz.

Signals from the Very High Resolution Radiometer are used to frequency modulate the C-band transmitter. The VCO used to generate the FM signal is extremely linear (±1%) in order to minimize distortion of the radiometer information.

In order to assist ground stations in directing the high-gain antennas toward the ATS spacecraft, a CW C-band radio beacon signal is transmitted in the medium gain mode.

A monopulse mode is implemented in the C-band frequency range for determining the attitude of the spacecraft in roll and pitch with an accuracy of about ±0.3 degrees. This signal can be fed directly to the attitude control system and to the telemetry system for transmission to ground stations.

The communication subsystem also provides C-band transmission for signals derived from the RFI Experiment.

Antennas

The antennas used with the communications system are located in two areas: (1) the prime focus feed, on the top surface of the EVM, which illuminates the 30-foot parabolic reflector, and (2) the earth viewing horns on the bottom surface of the EVM that receive and transmit signals directly to and from the earth. Figure II-26 shows the antenna layout on the top surface. Figure II-27 shows the bottom view of the subsystem.

Physical Description

The components that comprise the communications subsystem are located in the uppermost section of the EVM. The transponder is packaged in nine major assemblies as follows:

Translator
L-Band Filter
S/L-Band Power Amplifiers
S-Band Monopulse
RFI Transponder
Synthesizer

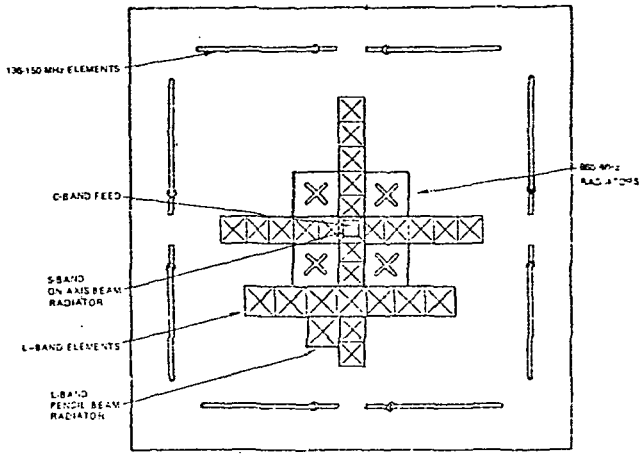


Figure II-26. Composite Antenna Feed

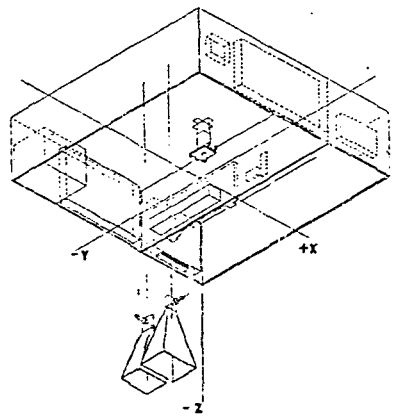


Figure II-27. Bottom View of Communication Subsystem Box Showing Transponder and Earth Viewing Horns

UHF Transmitters
C-Band Power Amplifiers
C-Band Switching and Diplexing

The Composite Antenna Feed is on the top surface of the communication module which locates it at the focal point of the 30-foot parabolic reflector.

Transponder

The transponder can be functionally divided into four major areas: the receiver, the IF Amplifier assembly, the frequency synthesizer, and the transmitter. Supporting these major areas are the RF input-output circuitry, wideband data unit, monopulse detector, command distributor, and dc-dc converters. Figure II-28 diagrams the signal interfaces.

The receiver includes preamplifiers, monopulse modulators, mixers, filter and the required interconnecting circuits.

The IF amplifier consists of an IF input switch matrix, three identical IF amplifiers (f_c = 150 MHz, Δf = 40 MHz or 12 MHz, commandable) and an IF output switch matrix.

The synthesizer uses a single frequency (100 MHz) oscillator from which it synthesizes all the other desired frequencies. It can operate in the coherent mode, wherein the VCO and all the frequencies derived from it are phase-locked to the received signal. In the noncoherent mode a highly accurate oscillator acts as the reference.

The transmitter portion of the transponder is comprised of the up-converter, the drivers and the power amplifiers for C-Band, S-Band and L-Band.

Tables II-11 thru II-17 give the signal interfaces, and the expected performance of the transponder.

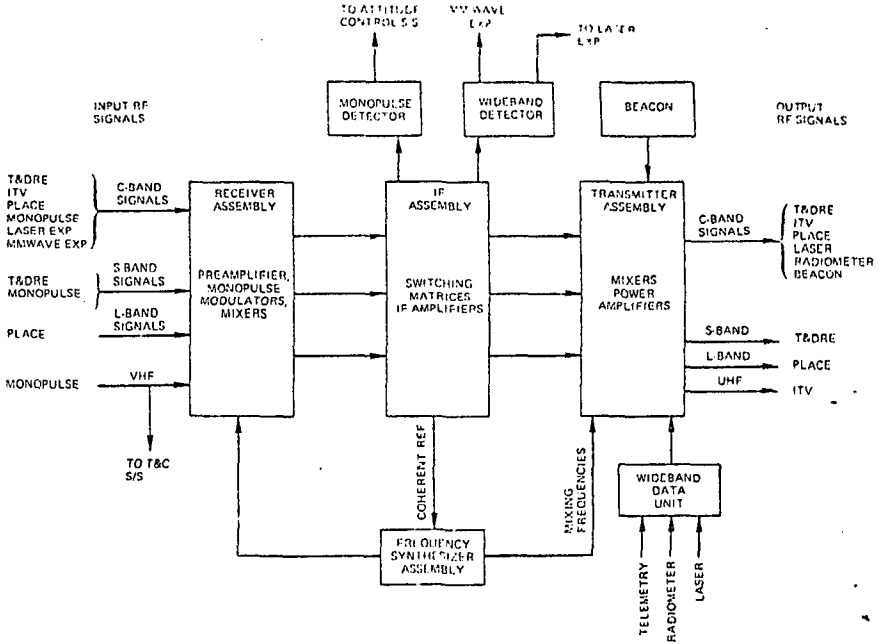


Figure II 28. Transponder Signal Interfaces

Table II-11
Transponder Signal Interfaces

Interface Name	Type	Frequency	Signal Level	Impedance
Laser Video Output	Experiment Output	0.1 Hz to 5 MHz	1 V P-P	50 ohms
RFI	Experiment Input	7 GHz	+13 dBm	50 ohms
T&FD	Experiment Input	1 ns pulse	TBD	TBD
C-Band Output	Signal Output	3750, 3950, 4150, MHz	20 W	50 ohms
S-Band Output	Signal Output	2060 MHz	21 W	50 ohms
L-Band Output	Signal Output	1550 MHz	40 W	50 ohms
UHF Output	Signal Output	460 MHz	50 W	50 ohms
Monopulse Output	Experiment Output	DC	±10 V	1-6 ohms
C-Band Input	RF Signal Input	6350, 6150, 5950, MHz	-40 to -55 dBm	50 ohms
C-Band Monopulse	RF Monopulse Sum and Error	6350, 6150, 5950, MHz	-90 dBm min	50 ohms
S-Band Input	RF Signal Input	2247, 2250, 2253 MHz	-86 to -115 dBm	50 ohms
S-Band Monopulse	RF Monopulse Sum and Error Signals	2247, 2250, 2253 MHz	-125 dBm min	50 ohms
L-Band Input	RF Signal Input	1651 MHz	-82 to -103 dBm	50 ohms
VHF Monopulse	RF Monopulse Sum and Error Signals	150 MHz	-55.5 dBm	50 ohms
Laser Video In	Experiment Video	0.1 Hz to 5 MHz	1 V P-P	75 ohms
Radiometer	Experiment Subcarrier	0.1 Hz to 150 kHz	1 V P-P	75 ohms
HR Camera	Experiment Subcarrier	200 kHz to	1 V P-P	75 ohms
Beacon	IF Input	130 MHz	-20 dBm	50 ohms

Table II-12
Expected Received Signal Levels

			Received Signal (Isotropic)		Transponder Input Signal[1]	
Mode	Band	Ant.	Max	Min	Max	Min
Coherent Receive	C	Horn Dish	-90 dBm	-105 dBm	-73 dBm -40	-88 dBm
PLACE	C to L L to C	Horn Fan Pencil	-75 -120	-100 -140	-55 -55 -82	-83 -103 -102
DRS	C to S S to C C to C	Horn Array Horn	-75 -125 -75	-100 -145 -100	-55 -86 -55	-83 -106 -83
Apollo Voice	S	Array	TBD	-157	TBD	-118
ITV	C to UHF	Horn Dish	-75	-100	-55 -25	-83 -50
MM, Laser, ATS-R	C to C	Horn	-75	-100	-55	-83
VHF Monopulse	VHF	Dish	--	-90	--	-72
Special Beacon	S to C	Array	TBD	-167	--	-125

Notes:

(1) Assumes the following nominal antenna gains:

C-band Horn	17 dB
C-band Dish	50
S-band Array	39
L-band Fan	32
L-band Pencil	38
VHF Dish	18

Table II-13
Frequency Synthesizer Performance Requirements

Phase Lock Loop Performance:

Noise Bandwidth (single sided)	500 Hz ± 10% nominal (1500 Hz max)
Minimum Capture Range	± 50 kHz at X-Band
Static Phase Error	2° RMS
Dynamic Phase Error	10° RMS
Acquisition Time	1 min (maximum)
VCXO Drift (open loop)	3 parts in 10^6 (maximum)

RF Input Requirements: 150 MHz IF at 220 mv signal + noise
Minimum signal/noise (IF BW = 40 MHz) = +1 dB

RF Output Requirements:

	Down Conversion	Up Conversion
Level	+3 dBm	+13 dBm
Spurious	-100 dBm	-60 dBm

DC Power Required

11 W ITV Mode
26 W all other modes

Table II-14
C-Band Performance

RECEIVE	
Transponder Input to Preamp Losses	
Sum Channel	
Diplexer	0.3 dB
TDA Switch	0.2 dB
Error Channel	
Modulator	1.0 dB
Filter	0.3 dB
Preamplifiers	
Noise Figure	5.5 dB
Gain	15.0 dB
Preamp to IF Losses	
Coupler	
Sum Channel	1.2 dB
Error Channel	7.0 dB
Triplexer	1.0 dB
Downconverter	5.0 dB
IF Switch	0.5 dB
IF Noise Figure	5.0 dB
Overall Receiver Noise Figure *	
Sum Channel and Horn	7.13 dB
Error Channel	9.5 dB
TRANSMIT	
TWTA Output Power (20 W)	+43 dBm
Switch Loss	0.25 dB
Coupler & Waveguide Loss	0.1 dB
Diplexer Loss	0.2 dB
Transponder Output Power	+42.45 dBm

*Includes input to preamp losses.

Table II-15
S-Band Performance

RECEIVE	
Transponder Input to Preamp Losses	
Sum Channel	
Filter	0.15 dB
Preamp Switch	0.2 dB
Error Channel	
Modulator	1.0 dB
Filter	0.15 dB
Preamplifiers	
Noise Figure	3.7 dB
Gain	34.0 dB
Preamp to IF Losses	
Coupler	
Sum Channel	1.2 dB
Error Channel	7.0 dB
Downconverter	7.0 dB
IF Switch	0.5 dB
IF Noise Figure	5.0 dB
Overall Receiver Noise Figure *	
Sum Channel & Cross Array	4.16 dB
Error Channel	4.93 dB
TRANSMIT	
Power Amplifier Output Power (21 W)	+43.2 dB
Switch Loss	0.2 dB
Cable Losses	0.2 dB
Transponder Output Power	+42.8 dBm

Includes input to preamp losses.

Table II-16
L-Band Performance

RECEIVE	
Transponder Input to Preamp Losses	
Diplexer	1.1 dB
Preamplifier	
Noise	4.4 dB
Gain	25.0 dB
Preamp to IF Losses	
Filter	0.1 dB
Downconverter	7.0 dB
IF Switch	0.5 dB
IF Noise Figure	5.0 dB
Overall Receiver Noise Figure*	5.57 dB
TRANSMIT	
Power Amplifier Output Power (40 W)	+46 dBm
L-Band TX/RCV Diplexer Loss	0.2 dB
Cable Losses	0.2 dB
Transponder Output Power	+45.6 dBm

*Includes input to preamp losses.

Table II-17
VHF and UHF Performance

VHF RECEIVE (150 MHz)	
Transponder Input Losses	
Sum Channel	
T&C diplexer	1.0 dB
T&C hybrid	3.0 dB
Coupler	1.0 dB
Error Channel	
Modulator	1.0 dB
Filter	0.2 dB
Coupler	7.0 dB
Both channels filter	1.6 dB
Line Switch	0.5 dB
IF Switch	0.5 dB
IF Noise Figure	5.0 dB
Overall Receiver Noise Figure*	
Sum Channel	12.0 dB
Error Channel	15.2 dB
UHF TRANSMIT (860 MHz)	
Power Amplifier Output Power (50 W)	+49 dBm
Filter Loss	0.2 dB
Cable Loss	0.2 dB
Transponder Output Power	+48.6 dBm

*Includes input to preamp losses.

TELEMETRY AND COMMAND SUBSYSTEM

The telemetry and command (T&C) subsystem receives, decodes and distributes commands transmitted from the ground to the spacecraft subsystems and experiments. The T&C subsystem also formats and transmits data associated with the spacecraft subsystems and experiments to the ATS ground stations. The subsystem block diagram is shown in Figure II-29.

Command signals may be received by either the VHF command receivers or the transponder C-band receiver.

The telemetry transmitters accept signals from any one of five sources selected by the data switching unit (DSU). These five sources are the data acquisition and control units (DACU), the environmental measurements experiment (EME), the transponder C-band receiver (in the special data mode), and the frequency division multiplex (FDM) unit. The FDM permits three sources simultaneous use of one transmitter connected to the prime focus feed.

The antenna and RF transmission system is configured to support the T&CS function from launch through all phases of the mission.

Command

The command section of the subsystem provides for reception of commands at the spacecraft and for separate command execution after verification by telemetry. The subsystem characteristics are listed in Table II-18.

All command functions may be performed using VHF or C-band uplinks. Simultaneous normal and ground attitude control (GAC) commands may be transmitted using one VHF uplink for normal commands and C-band for GAC commands.

Command functions performed by redundant T&CS equipment are as follows:

1) Receiving ground station command transmission

2) Processing normal commands, designated as discrete (relay) and data word commands

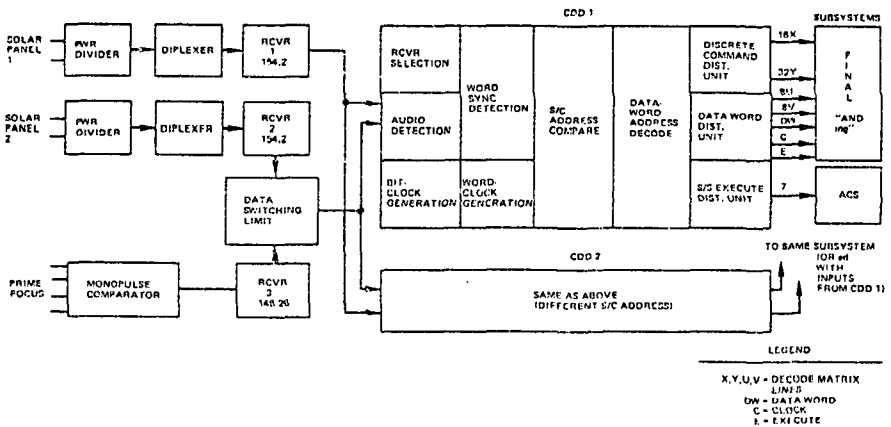


Figure II-29. ATS-F&G Telemetry & Command Subsystem

Table II-18
Command Subsystem Characteristics

Modulation: PCM/FSK/AM/AM at VHF
PCM/FSK/AM/FM at C-Band

Frequencies: 154.20 MHz, Prime
148.26 MHz, Standby
C-Band, Standby

Antennae: 2 Solar Panel Omnidirectional;
VHF Prime Focus Feed;
C-Band Earth Coverage Horn.

	COMMAND MODES	
Characteristic	Normal Command & Execute	GAC Command
Verification	Ground, via TM	Decoder address
Execute	Digital command	Execute as received
FSK Tones	Logic "0" 7296 Hz Logic "1" 7698 Hz	Logic "0" 8,760 Hz Logic "1" 11,560 Hz
Bit Rate	128 bps	1200 bps
Decoder Address	9 bits	7 bits
Frame Length	28 bits	13 bits
Discrete Commands	512	---
Data Word Addresses	60	---
Data Word Length	9	---
ACS Configuration Addresses	---	32
Normal Command *Execute Options*	a) Execute & clear b) Execute & hold	High speed execute of normal command
Execute Pulse Width	a) 50 ms, nominal b) 250 ms, nominal	11 ms, nominal

Table II-18
Command Subsystem Characteristics (Cont)

Characteristic	COMMAND MODES	
	Normal Command & Execute	GAC Command
Max. Command Rate	2/sec	92.3/sec
Repetitive Execute Rate	4/sec	92.3/sec
Time Duration Command Resolution	250 ms	11.25 ms
Command Distribution	a) Array for support subsystems b) Dedicated to experiments c) Remote distributor for transponder	Dedicated to ACS only
Redundancy	Redundant antennas, receivers & decoders	Redundant antennas, receivers & decoders

*Without ground verification before execute.
2.2/sec for 50 ms commands, 1.4/sec for 250 ms commands. Any rate up to 1.4/sec for either execute pulse width; non between 1.4 and 2.2/sec. Standard format rate required is 1/sec.

3) Decoding, storing, and telemetering valid normal commands, including data word parity, for ground station verification before execution

4) Decoding valid normal execute commands and signaling the addressed experiment or support subsystem interface

5) Processing valid GAC commands designated: APS thruster (or ACS wheels) configuration commands, high speed execute (repetitive, or time duration) commands, and Polaris pointing commands in near real-time without prior ground verification via telemetry

6) Providing in-band simplex VHF repeater connections for the special data link (SDL).

The command word consists of a 64-bit field and the command rate is two commands (128 bits) per second. The pulse duration is 50 milliseconds. There are 511 discrete commands and 63 command addresses that can be used.

The command logic levels are

-5 to 0 ± 0.5 volts for logic "0" or "off"
+5 ± 0.5 volts for logic "1" or "on"

The associated ground stations have an EIRP of 48.5 BW with an antenna beamwidth of 20 degrees. In the normal command mode the primary frequency (154.20 MHz) is used for all functions; i.e., simultaneous reception of FSK commands and 7-tone SAPPSAC commands. Receivers #1 and #2 connected to the omni antennas are tuned to this frequency. The secondary command frequency (148.26 MHz) will be used for special data mode operation on receiver #3, which is connected to the high gain antenna.

The VHF carrier is FSK modulated by the 0 and 1 command tones. The signal is 50% amplitude modulated by a 128-Hz sine wave that is demodulated in the command decoder/distributor (CDD) and used for all timing within the CDD and for data word transfer to the subsystems. Two CDD's are used, each with its own spacecraft address, so only one is used at any time. In the primary mode, the CDD's receive command signals from receivers #1 and #2. The CDD alternately samples the received signal strength and locks onto the receiver that first reaches a predetermined threshold level.

For the SAPPSAC experiment seven tones are used to amplitude modulate the carrier and are decoded by the CDD for timed-tone control of the ACS jets and wheels. Six tones are used for attitude control in three axes. The seventh is a holding tone that must be present during the entire timed-tone mode of operation.

The normal commands and the 7-tone mode may be frequency division multiplexed on the primary carrier for simultaneous operation. A third, voice bandwidth AM, signal can be transmitted on the secondary carrier frequency. This signal is switched by the data switching unit (DSU) to a telemetry transmitter for retransmission to the ground via the high-gain antenna. This mode of operation provides a simplex push-to-talk voice channel between ground stations to support the experiments.

Telemetry

The ATS-F and -G telemetry subsystem provides the means for monitoring spacecraft and experiments data during ground testing, prelaunch, launch and all orbital operations. The telemetry rate is 400 bits per second in a fixed format of gibit words with commandable dwell capability. Analog sampling is converted to a 9-bit digital word with an overall accuracy of ± 0.15 percent of full scale. The system provides a word length of 9 bits from the combination of analog, 9 bit serial digital or single bit (on-off) data inputs. The NASA 36-bit time code, converted to a PCM format is included in the telemetry data. Figure II-30 is a block diagram of the telemetry subsystem.

Analog input signals have a full-scale range of 0 to 5.12 volts with a data source impedance of less than 1k ohm.

Digital data source impedance is less than 10k ohms with the following logic levels:

-2.0 to +1.0 volts for logic "0"
+5.0 to +8.0 volts for logic "1"

The telemetry system transmits to SATAN and SCAMP ground stations on phase-modulated carriers at 136.23 and 137.11 MHz with a bandwidth restricted to 30 kHz. The ground antennas have a gain of 21 db and a 15 degree beam width.

In addition to its normal functions, the system also provides for transmission of the Environmental Measurements Experiment (EME) at a rate of 2000 bits/second simultaneously with telemetry data. It also provides a simplex communications link for the Special Data Link (SDL) with a 2.5-kHz basebank.

The telemetry subsystem includes four phase-modulation transmitters (2 watts each). Transmitter #1 (136.23 MHz) and transmitter #2 (137.11 MHz) are each connected to an omnidirectional antenna. Transmitter #3 (136.23 MHz) and transmitter #4 (137.11 MHz) can be individually selected and transmit through the prime focus and the 30-foot reflector. The capability of simultaneous transmission of telemetry data, EME data and the SDL is made possible by frequency division multiplexing. FDM permits combining three data channels on one telemetry transmitter driving the prime focus feed.

The data acquisition and control unit (DACU) performs the baseband time division multiplexing function. It accepts parallel, serial and on-off (single bit) digital data and analog data from the spacecraft subsystems and experiments.

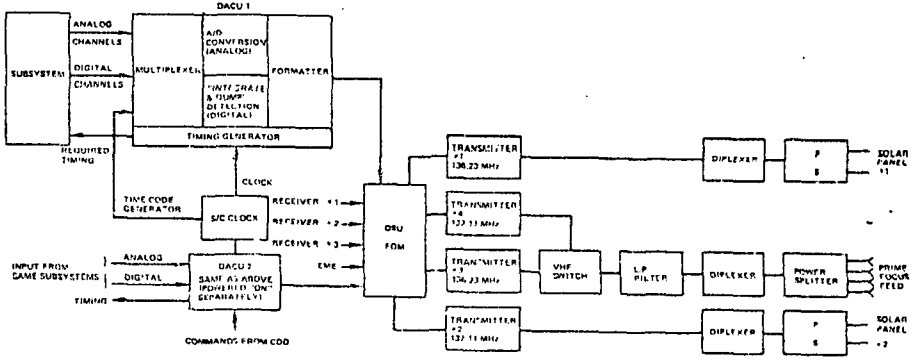


Figure II-30. Telemetry Subsystem Functional Block Diagram

It digitizes each analog sample in a 9-bit A-D converter. It sequences each telemetry word into its preassigned position in the format. The format consists of a 128-word minor frame with 16 minor frames per major frame. Basic timing for the DACU is derived from a 100.0 kHz spacecraft clock.

Table II-19 lists the subsystem parameters.

Table II-19
T & CS Telemetry Parameters

CHARACTERISTICS	ATS F&G
Frequencies	136.23 MHz, 137.11 MHz
Spacecraft Antenna	Omnidirectional mounted on solar panels. Directional via prime focus feed and parabolic reflector.
Modulation	PCM/PM on omni-associated transmitters. PCM FM PM or PM on prime focus feed-associated transmitters
Formatting	Data format Manchester II + 1F0 Fixed format with dwell capability
Bit Rate	390.625 nominal.
Word Length	9-bits
Minor Frame Length	128 words
Minor Frame Period	Approximately 3 seconds
Subcommutation	Last 16 words, 16 deep
Total Format Capacity	368 nine-bit words. Each word carries one digitized analog telemetry point or nine digital telemetry points.
Analog Channels	276
Digital Data	765 bits (55 nine-bit words)

REFLECTOR

The Flexrib reflector used on the ATS-F and -G provides a rugged, compact launch configuration that, through the use of simple release and deployment mechanisms, insures a very high probability of successful deployment in orbit. (See Figure II-31.) The parabolic reflector when deployed provides a gain of 49.0 db at 6 GHz (the C-Band receive frequency) and 46 db at 4 GHz (the C-Band transmit frequency).

The reflector consists of a reflecting surface formed by a copper-coated Dacron mesh supported by 48 radial thin-gauge aluminum ribs hinged to a central hub. The mesh, coated with silicon to protect the copper surface while in the terrestrial environment, has an optical transparency of about 71 percent coupled with a high reflectivity at the operating frequencies. The width of the ribs is tapered approximately 2:1 from root to tip. The thickness of the ribs is also tapered. The mesh is sewn to the edge of the ribs, which have been precision machined to the required contour.

For storage during launch the ribs are folded 90° at the hinges into a tangential position. They are then wrapped around the hub into the stowage container with the mesh carefully folded between the ribs. The swing doors are folded down over the reflector and held in place by an encircling cable. The total package, which weighs 140 pounds, forms a torus 12 inches high with an inside diameter of 58 inches and an outside diameter of 78 inches. The deployed reflector has a ratio of focal distance to diameter (f/d) of 0.44.

HARNESS

The function of the harness subsystem is to provide electrical connections for both signal and power to the equipments located in the communications module, the service module, the experiment module and the reflector hub. It also connects to the solar array, the separation and deployment pyrotechnics, the VHF antennas and the equipment mounted on the reflector truss. Table II-20 summarizes the harnesses by function. Figures II-32 through II-35 depict the configuration and placement of the harnesses.

The EVM harness layout in each of the three structural modules utilizes the north-south faces and interconnecting panels for cable routing. In the service module a fiberglass raceway for the harness is mounted on the edge of the center panel. In the communications module, the cables are clamped directly to the surfaces of the north, south and internal transverse panels.

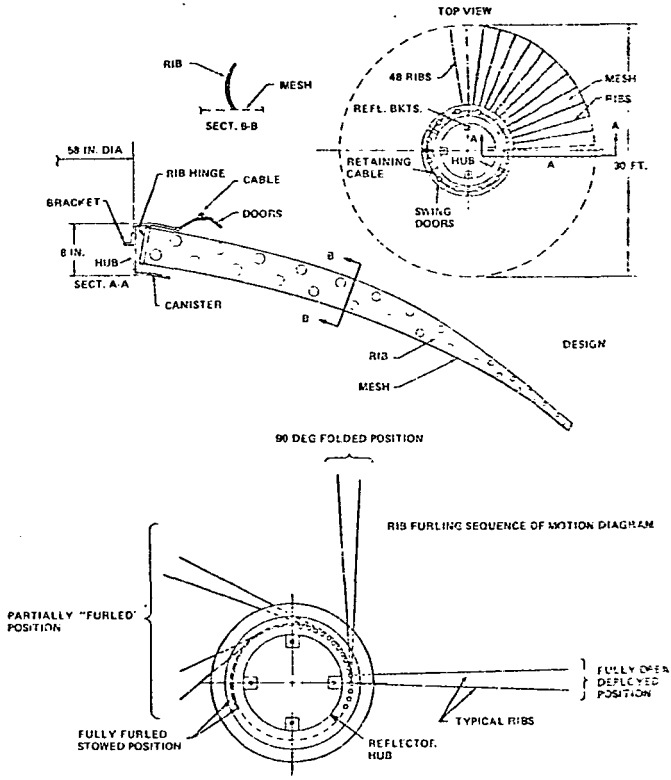


Figure II-31. Flexrib Reflector

Access to the connectors in the fully assembled spacecraft is provided by means of removable east and west panels.

Interface between the communications module and the service module is made through bulkhead connectors mounted on a plate attached to the lower edge of the communication module.

In the service module, the primary cabling is mounted near the upper edge of the north and south faces and of the interconnecting transverse panel. Interconnection between the experiment and service modules is made through bulkhead connectors located at the module interface and is accessible by removing east and west panels.

The external harnessing services the functions located on the reflector support truss, the reflector hub and the solar array deployment arms. The cabling to each area is constructed as a discrete functional harness and each is clamped to one of the reflector support truss members. Cables for equipment mounted on the reflector hub pass through the central section of the hub and mate with the appropriate components. The cables to the solar array and the VHF omni antenna also pass through the center of the hub and then out along the solar array booms.

Table II-20
ATS Harness Summary

FUNCTION CONSIDERED	WEIGHT ALLOCATED	RUN LENGTH	NUMBER OF WIRES	TOTAL WIRE LENGTH REQUIRED	WIRE TYPE	WEIGHT [illegible]	
Telemetry Monitor Lines	17.0 Pds.	5 ft. average	1500	7500 ft	24 AWG Single Shield Jacketed	3.6 Pds.	
Command Lines (system)	9.0 Pds.	5 ft. average	900	4550 ft	24 AWG Single Conductor Jacketed	2.0 Pds.	
Power Lines (system hookup)	3.5 Pds.	5 ft. average	70 unshielded 10 shielded	350 ft 100 ft	20 AWG Twisted Pair Jacketed 20 AWG Twisted Shielded Pair-Jacketed	11.2 Pds. 13.7 Pds.	
Attitude Control Subsystem Interconnection	6.5 Pds.	1 ft. average	650	2200 ft	22 AWG Single Conductor Jacketed	3.0 Pds.	[illegible]
Telemetry and Command Subsystem Interconnection	3.0 Pds.	1 ft. average	150 21	600 ft 193 ft	22 AWG Single Conductor Jacketed Coax	3.9 Pds. 11.2 Pds.	
Communications Subsystem Interconnection	9.0 Pds.	5 ft. average	450	1550 ft 270 ft	22 AWG Single Conductor Shielded Coax	4.6 Pds. 11.2 Pds.	
Power Subsystem Interconnection	3.0 Pds.	1 ft. average	50 unshielded 15 shielded	225 ft 75 ft	20 AWG Twisted Pair Jacketed 20 AWG Twisted Shielded Pair Jacketed	11.0 Pds. 13.7 Pds.	
Pyrotechnics Lines	2.0 Pds.	4 ft. average	50	200 ft	20 AWG Twisted Shielded Pair Jacketed	13.7 Pds.	
Solar Array Lines	7.5 Pds.	10 ft. average	16- power 4- Telemetry	640 ft 160 ft	20 AWG Twisted Pair Jacketed 24 AWG Twisted Shielded Pair-Jacketed	11.7 Pds. 7.4 Pds.	
EME Lines	2.5 Pds.	20 ft.	35 6 2	700 ft 120 ft 40 ft	24 AWG Single Conductor Jacketed 20 AWG Twisted-Shielded Pair Jacketed Miniature Coax	2.5 Pds. 13.7 Pds. 11.0 Pds.	
Communicable Gravity Gradient Lines	4.0 Pds.	20 ft.	10 4	200 ft 80 ft	24 AWG Single Shielded Jacketed 20 AWG Twisted Shielded Pair Jacketed	3.6 Pds. 13.7 Pds.	[illegible]
Sun Sensor and Amplifier	1.0 Pds.	20 ft.	4 5	80 ft 100 ft	22 AWG Twisted Shielded Pair Jacketed 24 AWG Single Shielded Jacketed	10.0 Pds. 3.6 Pds.	
VHF-OMNI Antenna Lines	4.0 Pds.	45 ft.	2	90 ft	Coax	11.0 Pds.	
Deployment Pyrotechnics	5.0 Pds.	30 ft.	15	450 ft	22 Twisted Shielded Pair Jacketed	10.0 Pds.	
Connectors and Accessories	11.0 Pds.	-----	-----	-----	-----	-----	[illegible]

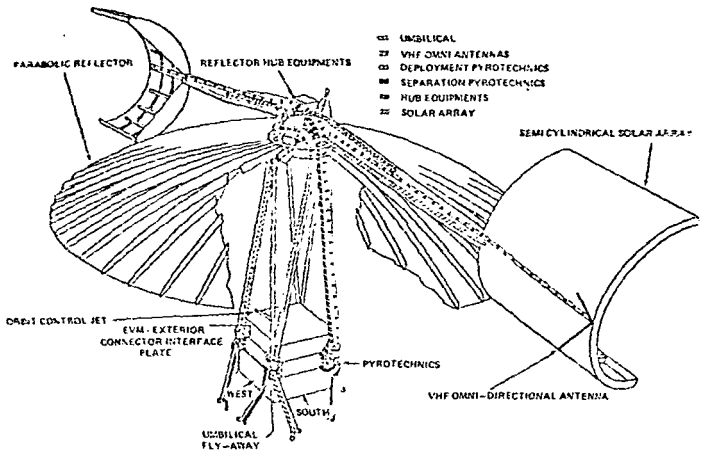


Figure II-32. ATS-F Spacecraft Configuration and External Harness

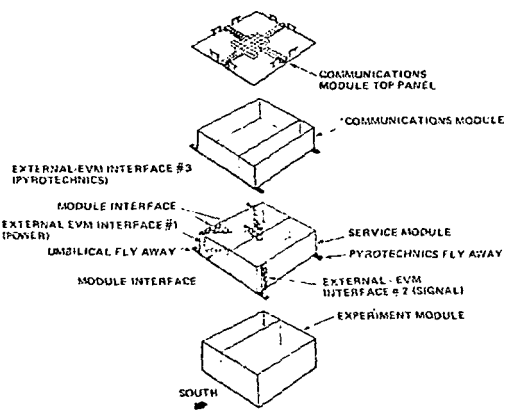


Figure II-33. Exploded EVM Connector Interfaces

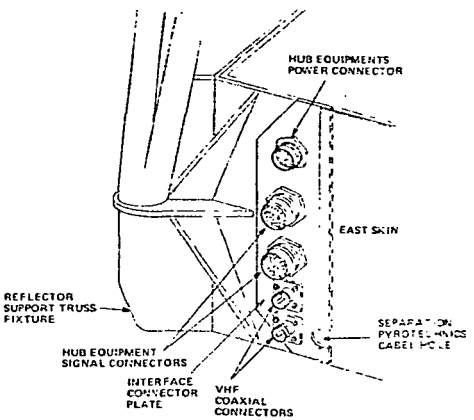


Figure II-34. EVM Interface

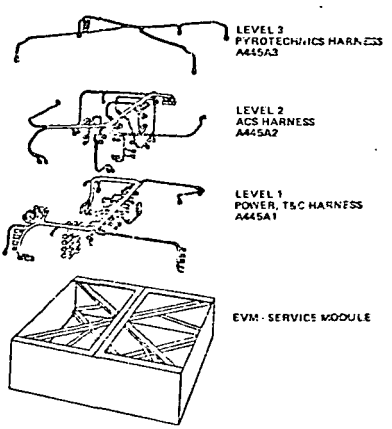


Figure II-35. Service Module Harnessing

RF INTERFEROMETER SUBSYSTEM

The RF Interferometer, when used in conjunction with two ground transmitters, provides the means of determining spacecraft attitude in roll, pitch and computed yaw to an accuracy of ±0.01S . This accuracy is attainable within a 12.5 conical field of view and ±0.025 at 30 conical field of view centered on the spacecraft Z-axis. (See Figures II-36 and II-37.) There are four commandable modes of operation:

- Resolution—Vernier/coarse-alternate
- Axis—Pitch/roll-alternate
- Frequency—f_1/f_2-alternate $f_1 + f_2$
- Calibrate—Phase reversal of baseline

A single ground station provides two-axis sensing (pitch and roll) while for three-axis sensing, two ground stations are required. In the single frequency mode, the interferometer may be directly connected to the attitude control system. In the baseline calibration mode, phase readings (phase and phase reversal) are taken over two diagonal baselines, thus providing in-flight calibration. Figure II-38 is a block diagram of the subsystem. Table II-21 summarizes the subsystem.

System Description

The RF Interferometer is comprised of four functional assemblies:

- Antenna Array—The antenna array, mounted on the earth-viewing surface of the EVM, consists of two orthogonal baselines, each baseline being orthogonal to the yaw axis of the spacecraft. The pitch baseline consists of three antenna elements along the pitch axis. The two elements spaced farthest apart (19.95λ) form the vernier baseline while the two close together (1.66λ) form the coarse baseline to resolve ambiguities. (See Figure II-39.)

- Receiver—A two-channel receiver is provided, with one channel for the reference signal and one for the comparison signal. A coupler/switch module provides for time division multiplexing between the signals from the coarse and vernier antennas. The first local oscillator down-converts the signals to 150 MHz. The signals are then applied to the second mixer where a 2 kHz frequency difference is introduced by the Dual Local Oscillator. Two signal paths, offset by 5 MHz, provide for simultaneous two frequency

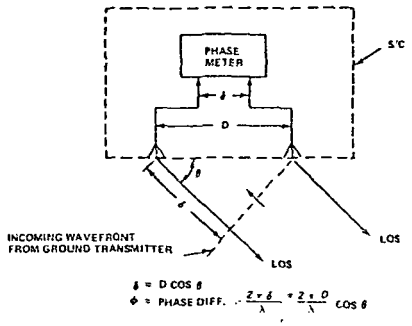


Figure II-36. Basic Equation Angle Measurement

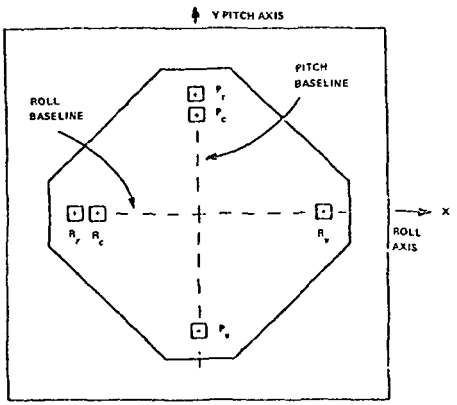


Figure II-37. Antenna Array Configuration

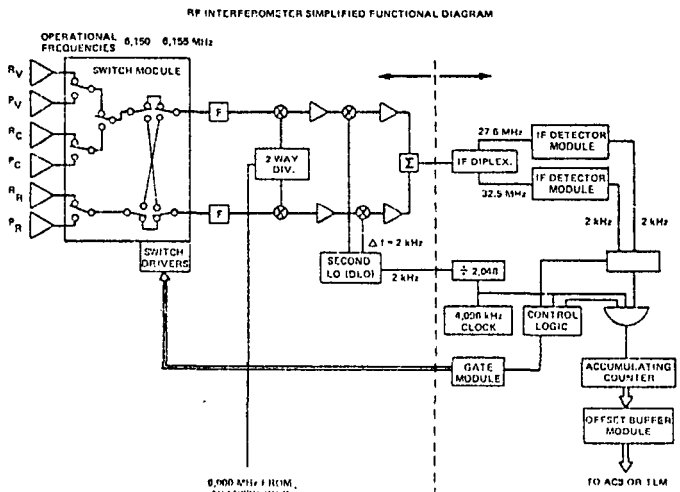


Figure II-38. Block Diagram of Interferometer Subsystem

Table II-21
R.F. Interferometer Subsystem Summary

Type of Receiver:	Dual conversion, dual channel Switched antenna elements
Input Frequency:	6.150 and 6.155 GHz
Noise Figure:	15 dB
Vernier Baseline:	38.292" (19.95λ)
Coarse Baseline:	3.191" (1.66 λ)
Configuration:	Crossed baselines with coarse element near end
Antenna Gain	+12 dB
Angle Coverage	±12.5° to±30°
Angle Accuracy (pitch, roll)	±0.18° (30), to±0.025° (55)
Antenna Element:	Compensated horn
First IF/IF Bandwidth (BW)	150 MHz/15 MHz
Second IF/IF BW	(Dual) 32.5 MHz and 27.5 MHz/1 MHz
Output Frequency/BW	2 kHz/ 750 Hz
First LO Frequency	6.000 GHz
Second LO Frequency	122.5 and 122.502 MHz
Post-Filter SNR	+41 dB at +73 dBW ground ERP
Vernier Clock Frequency:	1024 kHz
Coarse Clock Frequency:	1024 kHz
Counter Input Rate:	2 kHz
Vernier Averaging:	64 samples
Telemetry Output	Digital 60 bits/3 sec (1 or 2 sta modes)
Weight	18.5 lbs
Power	15.5 Watts

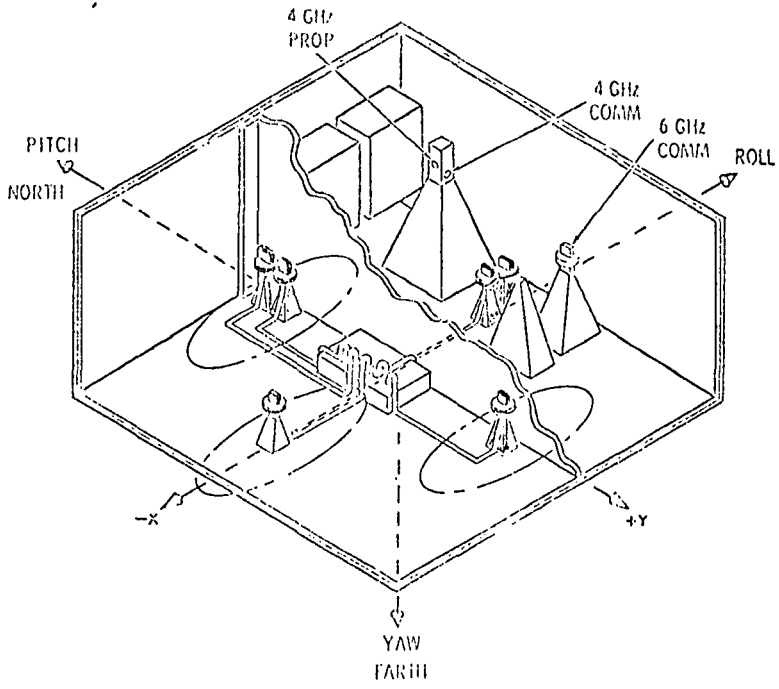


Figure II-39. Antenna Array Installation

operation. The outputs of the receiver system will be low frequency signals (2 kHz) which have preserved the phase relationship of the microwave signals received at the antennas.

- Spacecraft Data Converter—The interferometer data converter measures the phase relationship of the receiver output signals with respect to a coherent reference signal and converts these measurements to digital form which can be telemetered to the ground or connected to the Attitude Control System. A complete measurement can be made every 230 milliseconds and telemetered once every three seconds.

- Calibration Oscillator—The calibration oscillator provides a low-level test signal, upon command, to calibration injection ports of the antenna switch module for calibration of the phase error at 6.150 and 6.155 GHz.

Tables II-22 through II-25 list the link parameters and performance requirements.

Table II-22

Ground Station to Interferometer Link Parameters

Transmitter ERP (frequency = 6150 MHz)	+76 dBW
Propagation Path Attenuation	-201 dB
Rainfall Absorption	-1 dB
Receiver Antenna Gain	+12 dB
Receiver Signal	-84 dBm
IF Bandwidth (1.0 MHz)	+60 dB
Receiver Noise Figure	+15 dB
Signal-to-Noise Ratio in IF	-15 dB
Margin Improvement	+3 dB

Table II-23
Mixer/Amplifier Module Performance Requirements

SIGNAL INPUT PORTS	
RF Input Channel 1	
RF Input Channel 2	
First LO Input	
DLO Input #1	
DLO Input #2	
SIGNAL OUTPUT PORTS	
Channel 1 IF Output	
Channel 2 IF Output	
RF INPUT (each channel)	
Frequency	6.150 MHz and 6.155 MHz
Power Level (linear operation)	-95 dBm to -60 dBm
OVERALL GAIN (each channel, combiner detached)	
RF Input to IF Output	40 dB, 1 dB
1 dB Gain Compression Point	0 dBm IF output power level
PHASE TRACKING	
(Between channels 1 and 2 over frequency range 6147.5 to 6157.5 MHz, over dynamic range, and over required temperature range as specified in 7916881)	5 maximum

Table II-23

Mixer/Amplifier Module Performance Requirements (con't)

GAIN TRACKING	
(Between channels 1 and 2 over frequency range 6147.5 to 6157.5 MHz, over dynamic range, and over required temperature range as specified in 7916881)	±1.0 dB maximum
OVERALL NOISE FIGURE (each channel)	10 dB design goal
FREQUENCY RESPONSE (each channel)	
-0.5 dB Bandwidth	10 MHz, ±0.5 MHz
-3 dB Bandwidth	15 MHz, ±0.5 MHz
Rejection	70 dB minimum
Rejection Bands	DC to 6.095 GHz 6.45 GHz to 18 GHz
ISOLATION	
Between LO Ports and RF Input Ports	60 dB minimum
Between channel 1 Input and channel 2 Output	60 dB minimum
Between channel 2 Input and channel 1 Output	60 dB minimum
Between LO Ports and IF Output Ports	60 dB minimum
VSWR	
All Signal Input and Output Ports	1.25: 1 max, 50 ohms nominal

Table II-23

Mixer/Amplifier Module Performance Requirements (con't)

LO INPUT PORT	
Frequency	6000 MHz, ±0.06 MHz
Input Power	+9 dBm (nominal)
DLO INPUT PORT	
Frequency	Channel 1 : f_1 = 122.500 MHz ±24 kHz Channel 2 : $f_2 = f_1 + 2.0$ kHz
Input Power (each channel)	+9.0 dBm, ±0.5 dBm
IF OUTPUT PORTS	
Frequency	27.5 MHz and 32.5 MHz nominal

Table II-24

Double Local Oscillator Module Performance Requirements

Number of RF Outputs	2
Output Frequency Port #1	f_1 = 122.500 MHz
Port #2	$f_2 = f_1$ + reference signal frequency
Output Waveform, each port (RF)	Sinewave
Output Power, each port (RF)	+10 dBm ±0.5 dB
Power Outputs Matched to Within	±0.5 dB
Isolation Between Outputs Ports	40 dB minimum
Output VSWR	1.20:1 maximum
Output Impedance	50 ohms nominal
Phase Stability (between 2kHz beatnote from the two RF outputs and the 2kHz reference input)	±5° maximum
Frequency Stability	
Long Term	±10 ppm/maximum drift
Short Term	Phase noise at the oscillator RF output ports shall be no greater than 0.2 degree RMS between 50-10000 Hz about the carrier
Input Reference Signal Characteristics	
Waveform	-0.5 to +5.5 volts maximum squarewave
Frequency	Squarewave, 2 kHz ±0.1%
Impednace	1000 ohms, nominal
Interferometer Telemetry Output	PLL unlocked -2.0 to +0.4 VDC PLL locked +2.4 to +5.0 VDC
Loop Acquisition Time	1 second, maximum
Power Dissipation	1.8 W maximum

Table II-25

If Amplifier Detector Module Performance Requirements

No. of Signal Inputs	2
No. of Signal Outputs	2 (2 kHz signal) 2 (analog to telemetry)
Input Signal Characteristics	
Signal Frequencies	27.5, 27.502 and 32.5, 32.502 MHz
Signal Level	-58 dBm to -15 dBm
Impedance	50 ohms, nominal
VSWR	1.3 maximum
Output Signal Characteristics at Each Port	
Frequency	2 kHz
Level	1.2 V (p-p), ±0.2 V over the input signal dynamic range
Impedance	1000 ohms, nominal
Phase Stability	±5° maximum error
Output Isolation (between channels)	50 dB minimum
Harmonic Suppression (2 kHz harmonics)	30 dB minimum
Spurious Response	60 dB minimum
Spurious Response outside of (1.5 kHz to 2.5 kHz)	30 dB minimum
AGC/Audio Filter Combined Response Time	6 msec maximum
Noise Figure	14 dB nominal
Bandwidths	
Predetection (-3 dB)	1 MHz minimum
(-60 dB)	10 MHz (maximum)
Power Dissipation	2.4 watts (maximum)

SECTION III

LAUNCH VEHICLE

TITAN III-C Description

ATS-F and -G Mission Specification

SECTION III

LAUNCH VEHICLE

TITAN III-C DESCRIPTION

The Titan III-C consists of a three-stage liquid propellant core vehicle supplemented by two solid rocket motor (SRM) strap-ons (see Figures III-1 and III-2). The complete four-stage launch vehicle (less payload and fairing) has an overall length of approximately 110 feet and a liftoff weight of 1.4 million pounds.

Stage 0

Stage 0 consists of two identical SRMs, mounted 180 degrees apart on the core vehicle. The SRMs lift the vehicle off the launch pad with a thrust of approximately 1.2 million pounds each. Each SRM is made up of a forward closure, an aft closure, and five identical motor segments and includes instrumentation, destruct devices, rocket start devices, staging rockets, and a thrust vector control (TVC) system. TVC is provided by liquid injection of pressurized liquid nitrogen tetroxide (N_2O_4).

Stage I

Stage I is of aluminum skin-stringer construction and consists of two liquid propellant tanks with the necessary skirts and two gimbaled engine assemblies. It is 10 feet in diameter and approximately 71.5 feet long.

Stage I contains a fuel and an oxidizer tank as an integral part of the airframe. On the aft end of Stage I are four longerons that provide attachment points for the SRMs, as well as for the liquid propellant engines. The hydraulic system, a rate gyro package, and the Stage I destruct system are located in Stage I along with various instrumentation sensing devices.

The Stage I propulsion system uses an Aerojet YLR87-AJ-11 engine assembly comprising two gimbaled thrust chamber assemblies and related equipment. The engines are pump fed and combined regenerative and ablative cooled. The normal vacuum thrust is approximately 520,000 pounds. Propellants are a blend of hydrazine with unsymmetrical dimethylhydrazine and nitrogen tetroxide (N_2O_4).

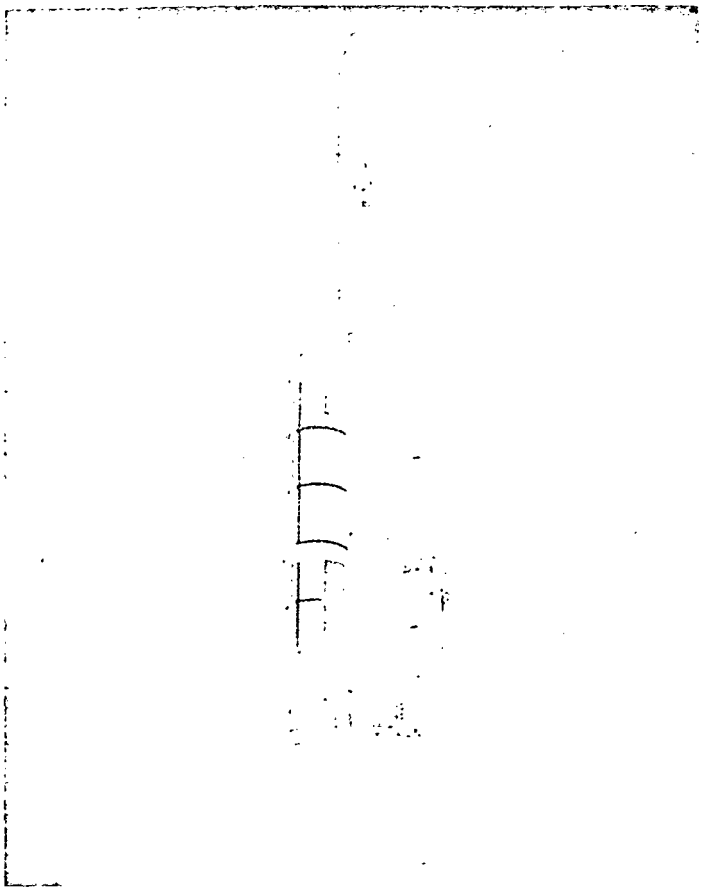


Figure III-1. Titan III-C Launch Vehicle

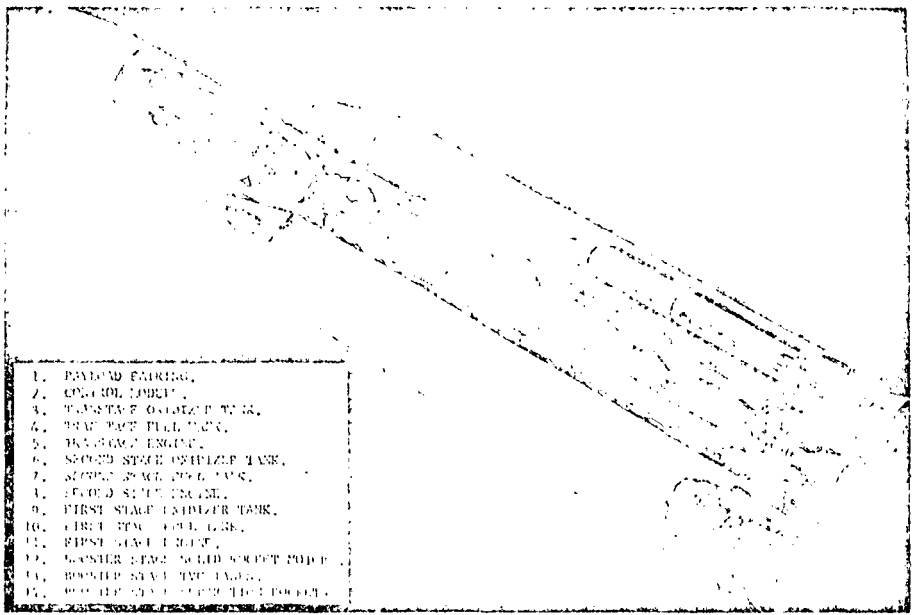

1. PAYLOAD FAIRING.
2. [illegible] SHROUD.
3. TRANSTAGE OXIDIZER TANK.
4. [illegible] FUEL TANK.
5. TRANSTAGE ENGINE.
6. SECOND STAGE OXIDIZER TANK.
7. [illegible] SKIRT.
8. SECOND STAGE ENGINE.
9. FIRST STAGE OXIDIZER TANK.
10. FIRST STAGE FUEL TANK.
11. FIRST STAGE THRUST
12. CENTER STAGE SOLID ROCKET MOTOR.
13. BOOSTER STAGE TVC [illegible].
14. BOOSTER STAGE [illegible] MOTOR.

Figure III 2. Titan III C Cutaway View

Stage II

Stage II is of the same type of construction as Stage I and consists of two liquid propellant tanks with skirts, between-tank truss, and one gimbaled engine assembly. It is 10 feet in diameter and approximately 31 feet long.

Stage II also comprises fuel and oxidizer tanks integral with the air-fram The engine is mounted to the aft cone of the fuel tank. The Stage II hydraulic system is located in the aft skirt of this stage. A rate gyro package, lateral accelerometer, the Stage II instrumentation system, command control receiver and the Stage III destruct system are located between the tanks.

The Stage II propulsion system uses an Aerojet LR87-AJ-11 engine assembly consisting of a single gimbaled thrust chamber and related equipment, and a gimbaled gas generator exhaust nozzle for roll control. It is pump fed with regenerative and ablative cooling. Normal vacuum thrust is approximate 100,000 pounds. Propellants are identical to those of Stage I.

Transtage

The transtage (Stage III) is 10 feet in diameter and approximately 14.5 feet long. It is of aluminum skin-stringer construction and consists of two liquid propellant titanium tandem tanks, two equipment trusses, and two gimbaled engine assemblies. Payload and fairing interfaces are provided at the forward end of the transtage at Vehicle Station 77.

The transtage consists of a propulsion module and a control module, connected by a manufacturing splice. The control module contains the inertial guidance subsystem, portions of the flight control subsystem, tracking and flight safety, instrumentation, electrical, and attitude control subsystems. The propulsion module contains the main transtage propulsion system, including propellant tanks, pressurization, propellant feed, and rocket engines. The main propulsion system provides the main impulse for payload maneuvering, while the attitude control system provides small velocity increments and attitude control.

The transtage main propulsion system uses an Aerojet AJ10-138 engine assembly consisting of two 8000-pound normal vacuum thrust subassemblies. The engines are ablative cooled and pressure fed using helium as a pressurant. Propellants are identical to those for Stages I and II. Multiple start capability is provided.

The transtage multipurpose attitude control system uses a monopropellant blowdown hydrazine system and fixed thruster assemblies to provide attitude control; propellant settling; orbit adjust, maneuvering, and vernier control; and multipayload deployment and controlled dispersion.

All electrical power required for the transtage payload fairing, and payload is derived from transtage-mounted silver-zinc storage batteries. Power is a nominal 28 vdc on five different buses and is provided from liftoff until payload separation (approximately 6.75 hours).

UHF Systems

Engine shutdown and destruct commands are supplied by two redundant independent UHF systems. Tracking is provided by a C-band pulse transponder and related equipment.

Hydraulics System

The hydraulics system is used in each of the three liquid propellant stages to gimbal the thrust chambers of the respective stage. In Stage II, the gas generator exhaust nozzle is also gimbaled for roll control. Electric valves are included in Stage 0 for TVC injectant and requires no hydraulics.

Guidance, Control, and Instrumentation Systems

The Titan III-C inertial guidance system (IGS) consists of a missile guidance computer (MGC) and an inertial measurement unit (IMU). The IMU is a derivative of the DE/CP Carousel IV IMU which is widely used on commercial aircraft. It is a four-gimbal system employing a unique design in which the x-y instruments are mounted on a rotating platform (see Figure III-3), thus minimizing the effect of inertial sensor errors in the x-y plane. The MGC, which is passively cooled, is a general purpose, parallel machine with a random access memory of sixteen thousand words of twenty-four bits each; fifty six instructions are available.

The flight control system stabilizes the attitude of the vehicle in all phases of flight launch through payload separation. This system establishes the flight path of the vehicle by implementing all steering commands issued by the IGS. It consists of software in the MGC, Stage I and II attitude rate sensors, lateral acceleration sensing system, TVC for the SRMs, and hydraulic actuators in the three core stages.

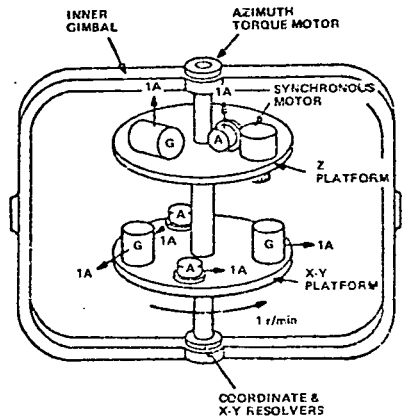


Figure III-3. Four-Gimbal System

The instrumentation system, a PCM/FM system, operates in the S-band frequency. Data signals are sampled and encoded by a remote multiplexed instrumentation system, which includes a group of remote multiplex units (RMU) and a single central converter. The RMU samples, amplifies, and holds the signals to provide a serial pulse-amplitude modulation (PAM) output train to the central converter.

Ordnance

Ordnance devices are used throughout the vehicle to provide prevalve opening, turbopump start, stage and fairing separation, SRM ignition, and vehicle destruct.

Fairing

The payload fairing design is based on a building-block concept where the fairing length can be tailored to the payload requirements with a minimum of fabrication effort. A 6-foot aft skirt section and a 9-foot nose cone/cone section comprise the basic 15-foot fairing length. Present plans for ATS-F and -G are to use a 30-foot nose fairing.

All sections of the fairing are fabricated from aluminum alloy materials. The aft 6-foot section and intermediate 5-foot modules are of semimonocoque construction consisting of thin skins, with 60 external stringers and internal ring-frames on 15-inch centers. The nose cone/cap section is thin skinned with internal ring-frames on approximately 6-inch centers. All sections of the fairing are assembled in longitudinal trisections at the factory using an internal bolted joint connection at the closeout frames of the sections.

Each section of the fairing contains three longitudinal cylinder-and-piston separation joints located 120 degrees apart. When the sections are assembled, three continuous separation joints are formed extending from the base to the top of the fairing. Each joint contains a separation device.

The fairing attaches to the transtage longerons at Vehicle Station 77.00 (transtage/payload interface). Compression loads are reacted on tapered bearing blocks located on the interface frame (or payload foot) directly above each longeron. An external tension fitting provides tension reaction from the fairing to the longeron bearing pads. Fairing shear loads are transmitted to the interface frame by fairing-mounted retractable shear pins that engage into holes in the frame. Both the compression and tension bearing surfaces on the fairing and transtage are beveled 5 degrees to facilitate disengagement during the lateral movement of the fairing.

Launch

The ATS-F and -G Titan III-C vehicles are currently scheduled to be launched from Launch Complex 40 at the Eastern Test Range. This complex is part of the Integrate-Transfer-Launch (ITL) facility designed especially for Titan III.

In the ITL concept, the liquid propellant stages are received, assembled vertically, and checked out in the vertical integration building (VIB) at the same time that the SRM segments are being assembled and checked out in the solid motor assembly building (SMAB). The core vehicle and SRMs are then joined in the SMAB and the entire launch vehicle is transported in the "ready" condition to the launch pad. Rail vans are used to house primary aerospace ground equipment (AGE) for the booster, and payload if required. The AGE in the vans is connected to the launch vehicles in the VIB and remains connected while being transported via the railway to the launch pad along with the vehicle. Although the umbilicals remain connected, electric power is generally not supplied to the vehicle while in transit between the VIB, SMAB, or launch site. Operations at the launch site (see Figure III-4) include combined system test exercises, range open-loop tests, liquid propellant servicing, tank pressuri-

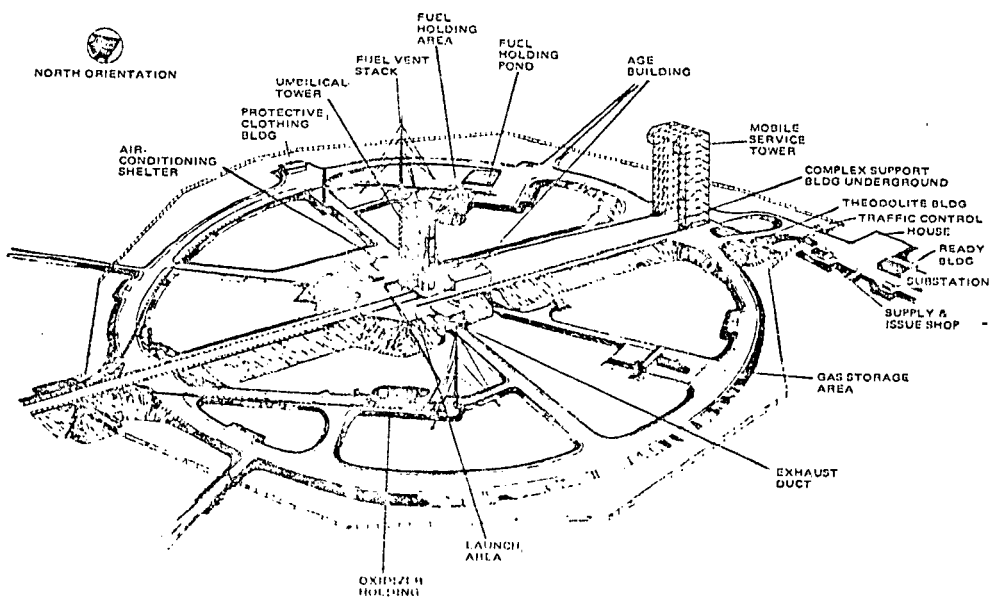


Figure III 4. Typical Titan III Launch Site

tion, vehicle checkout, and launching. The ATS-F and -G spacecraft will be mated to the launch vehicle at the launch pad.

Launch control centers are also located at the VIB in space provided above the low-bay area. These control centers consist of launch control rooms, payload control rooms, and the instrumentation/data transmission rooms. The control center contains consoles, support equipment, and instrumentation required for remote control and monitoring of launchings, launch countdowns, and flight operations. The payload control rooms are available for payload remote control, monitoring, and instrumentation equipment. The instrumentation/data transmission rooms contain ground instrumentation equipment and data links to the AGE vans, either in the VIB cell or at the launch pad.

ATS-F AND -G MISSION SPECIFICATION

ATS-F and -G launches will take place from pad 40 (AFETR) at a 93 degree flight azimuth. The spacecraft and transtage will be injected into a nominal elliptical parking orbit. At the second equatorial crossing (first crossing in a northerly direction) the transtage engines will be ignited and a transfer maneuver will be performed. The transfer maneuver will simultaneously produce an orbit with apogee at a nominal synchronous altitude and reduce the orbital inclination by approximately 2.25 degrees. At first apogee of the transfer orbit, the transtage engines will be ignited for the second time and a plane change/circularization maneuver will be performed to produce a nearly synchronous-circular orbit with an orbit inclination of 2.0 degrees, maximum. Following injection into the final orbit, the spacecraft will be separated and the transtage will perform a terminal maneuver (retro maneuver .

SECTION IV

ATS OPERATIONS

Introduction

ATS Operations Control Center

NASA Communications Network

Computation Division

Information Processing Division

SECTION IV

ATS OPERATIONS

INTRODUCTION

The primary responsibility of ATS operations is to ensure that the objectives of the mission are accomplished. The operational responsibilities involved in achieving the mission include—

- Ensuring spacecraft safety
- Controlling the flight mission
- Monitoring and evaluating spacecraft events
- Conducting experiments
- Providing orbit and spacecraft computations
- Processing and disseminating telemetry data

Major elements of the ATS operations ground systems are—

- Four ground stations
- ATS Operations Control Center (ATSOCC)
- NASA Communications Division (NASCOM)
- Trajectory Analysis Division
- Information Processing Division

The four ground terminals consist of three principle ground stations and the mobile terminal. The three principle ground stations, all having similar equipment, are—

- Rosman (near Brevard), North Carolina
- Mojave (near Barstow), California.

- Transportable[1], located at the Mojave site.

All four stations have—

- Parabolic antennas
- C-, S-, and L-band frequency modulated (FM) transmitters
- C-, S-, and L-band (FM) receivers.

For the ground station capabilities, see Section IV.

In addition to the above major stations, there will be a separate terminal for the millimeter wave experiment. For a complete description of the ground stations, see Section VI.

The instructional television experiment (ITV) will be conducted by the Indian Government in cooperation with NASA. Three Indian stations will originate TV programs for relay through ATS-F. These are Ahmedabad, Bombay, and Delhi (for details, see Section VII, Technological Experiments).

ATS OPERATIONS CONTROL CENTER

Responsibilities—The ATS Operations Control Center serves as the central point for the scheduling and coordination of all ATS communications and scientific experiments. The ATSOCC—

- Maintains optimal control of the spacecraft
- Coordinates with participating agencies
- Directs operations at the ground stations
- Collects R&RR and other data as required for determination of orbit and spacecraft attitude
- Schedules experiments and maintains data plots

[1]The Transportable Ground Station will be moved sometime after February 1972 to support ATS-F experiments.

- Disseminates orbital predictions and other data as required.

The flow of a large amount of data is required to achieve the objectives of the ATS project (see Figure IV-1).

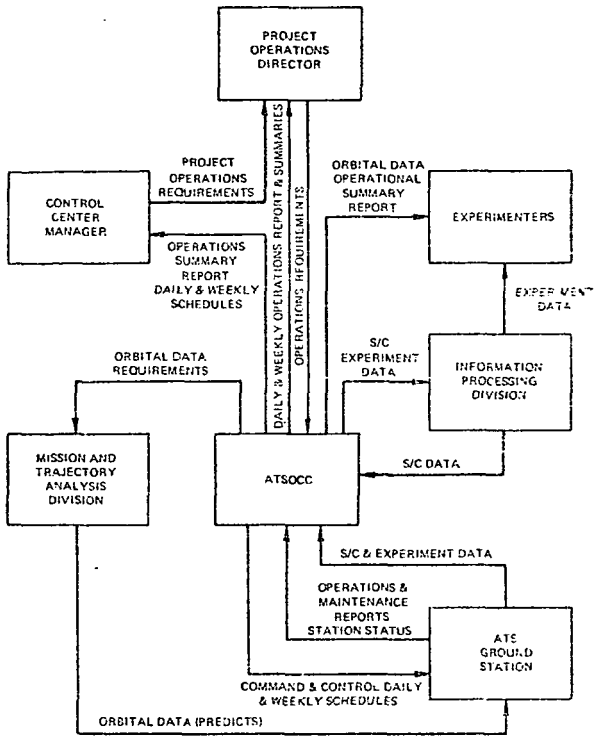


Figure IV-1. ATSOCC Operational Data Flow

NASA COMMUNICATIONS NETWORK

The NASA Communications Network (NASCOM) is a worldwide long-line communications system linking all tracking, data acquisition and command stations to GSFC. The ATS-F and -G project uses only a small portion of this vast network as shown in Figure IV-2.

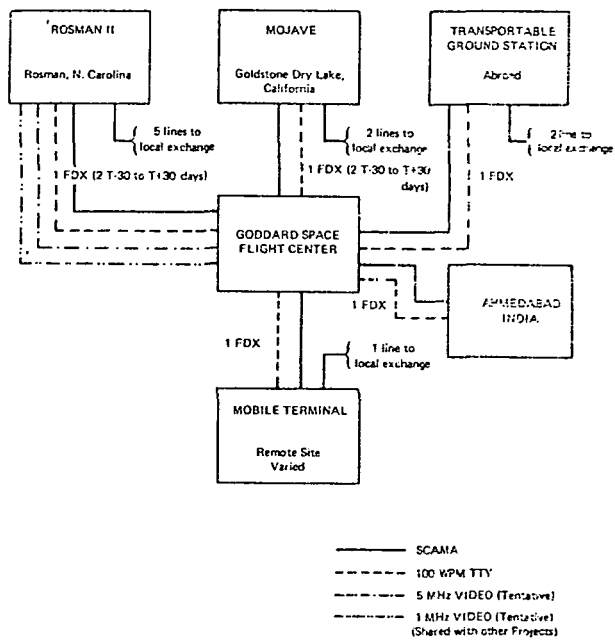


Figure IV-2. ATS Communications Landlines

TRAJECTORY ANALYSIS

The Computation Division is responsible for—

- Receipt and processing of tracking data
- Orbit and attitude determination
- Computations for spacecraft maneuvers
- Production of ephemeris data for experimenters

INFORMATION PROCESSING DIVISION

The Information Processing Division (IPD) is responsible for processing the ATS-F and -G telemetry data recorded by the ATS ground stations. IPD responsibilities include—

- Evaluation of the station tapes
- Conversion of the data from analog to digital
- Decommutation
- Shipment of experiment tapes to the experimenters

Data analysis of the experiments is performed by the individual experimenters.

SECTION V

TRAJECTORY AND ORBIT

Description of Maneuver

Description of Control System

SECTION V

TRAJECTORY AND ORBIT

DESCRIPTION OF MANEUVER

Launch Sequence

ATS-F (and later ATS-G) will be launched from the Eastern Test Range aboard a Titan III-C launch vehicle on an azimuth of 93° East. The spacecraft will be placed into near synchronous equatorial orbit at 94° West longitude.

The flight of the Titan III-C begins with the vehicle liftoff and ends with the separation of the spacecraft from the transtage. Following separation, the transtage is reoriented so that it may perform a retro maneuver to remove it from the path of the spacecraft to preclude any subsequent interference with ATS orbit operations.

A typical flight profile consists of a boost phase and an orbit phase.

Boost Phase--The boost phase covers the burning time of Stage zero and the operating times for the Stage I and Stage II engines. The significant events that occur during the boost phase are listed in numerical order and correspond to the numbered sequence of Figure V-1 and V-2.

1. Start solid rocket motors
2. Thrust decay
3. Stage I engine ignition
4. Staging of solid rocket motors
5. Stage I shutdown staging--Stage II ignition
6. Fairing separation
7. Stage II shutdown—staging

Orbit Phase—The orbit phase events are:

8. Fire attitude control system (multiple, as required)
9. Stage III ignition (multiple)
10. Payload separation.

Following the coast of the spacecraft to synchronous altitude, the transtage will thrust for about 120 seconds to inject the spacecraft into a nominal synchronous near equatorial circular orbit. Following the injection, the transtage

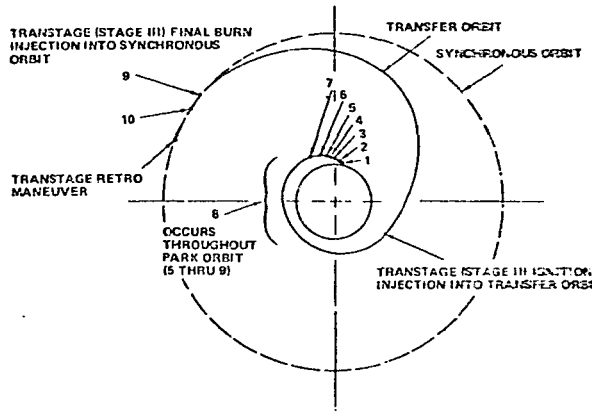


Figure V-1. ATS-F and -G Mission Profile

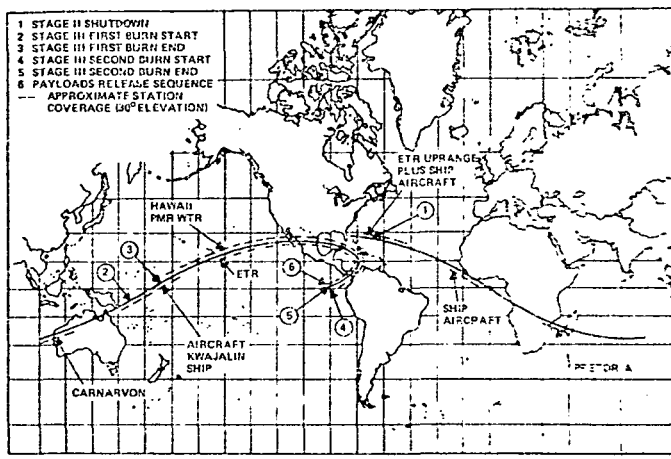

1 STAGE II SHUTDOWN
2 STAGE III FIRST BURN START
3 STAGE III FIRST BURN END
4 STAGE III SECOND BURN START
5 STAGE III SECOND BURN END
6 PAYLOADS RELEASE SEQUENCE
--- APPROXIMATE STATION COVERAGE (10° ELEVATION)
ETR UPRANGE PLUS SHIP AIRCRAFT
HAWAII PMR WTR
ETR
SHIP AIRCRAFT
AIRCRAFT KWAJALIN SHIP
CARNARVON

Figure V-2. ATS-F and -G Nominal Ascent Trajectory

provides signals and power to the spacecraft separation ordnance separate the spacecraft from the transtage. A backup capability by ground command is provided from the Rosman Ground Station.

The spacecraft is injected from the parking orbit into the transfer orbit at the second equatorial crossing with the apogee at an altitude of approximately 19,323 nm.

The spacecraft/transtage coasts for about 5.3 hours in the transfer orbit. During this coast, the transtage will perform thermal maneuvers for the spacecraft. This ensures proper operation of the spacecraft thermal control system during the transfer orbit. The spacecraft solar panels will be aligned with the sun line to enhance solar power generation since elements of the Attitude Control Subsystem (ACS) and the Telemetry and Command Subsystem must be powered during the launch phase.

Acquisition Sequence

With the spacecraft now in synchronous orbit, the angular rates caused by the separation must be reduced to less than 1.0 degree per second so that deployment of the solar panels and the 30-foot parabolic reflector may be accomplished. A command is sent from the Rosman Ground Station to reduce the angular rates about the three spacecraft axes to less than 1.0 degree per second. After successful deployment of the solar panels and the 30-foot reflector (by command) and when the spacecraft body angular rates have become less than 0.05 degree per second, the acquisition of the sun begins. This step starts from any initial spacecraft attitude using the 4-π steradian field of view of the coarse sun sensors. The 4-π steradian field of view brings the sun within the 9-degree acquisition field of view of the fine sun sensor. The spacecraft X-axis is thus brought within one degree of the sun by pitch and yaw control. When the sun acquisition is completed, the spacecraft is caused to roll about its +X axis. The roll axis control is automatically switched to the earth sensor when the earth sensor roll head indicates the presence of the earth. The +Y (pitch) axis control is switched to the earth sensor when the roll and pitch head indicates earth acquisition (both horizons are in the field of view). Earth acquisition is complete when pitch and roll errors are less than 1.0 degree.

The next step is the acquisition of Polaris. Commands are sent to select star intensity threshold settings and to offset roll. The roll-offset command defines the position of Polaris so that it can be acquired by a yaw angle maneuver using the inertial reference assembly as the reference sensor. Control of the +Z (yaw) axis is transferred to the Polaris sensor when the yaw

error is less than 2.0 degrees and the presence of Polaris is indicated by the Polaris sensor. Polaris acquisition is complete when the yaw error is less than 0.5 degree of the line-of-sight of the star.

With the ACS now nulling the roll and pitch outputs from the Earth sensor and the yaw output from the Polaris sensor, the spacecraft is maintained in a local vertical orientation.

Operational Modes

When ATS-F has been placed in initial orbit, the satellite ephemeris data is sent to the digital operational controller (DOC). Using this data, computation is made to correct the orbit to the desired circular synchronous orbit. A command is then sent to the spacecraft to activate the orbit-control mode.

So that the spacecraft may perform earth pointing, low altitude satellite tracking, and attitude and orbit control maneuvers, the following spacecraft attitude control modes are used to acquire the reference orientation and to maintain the spacecraft's attitude relative to this reference coordinate system (Figure V-3).

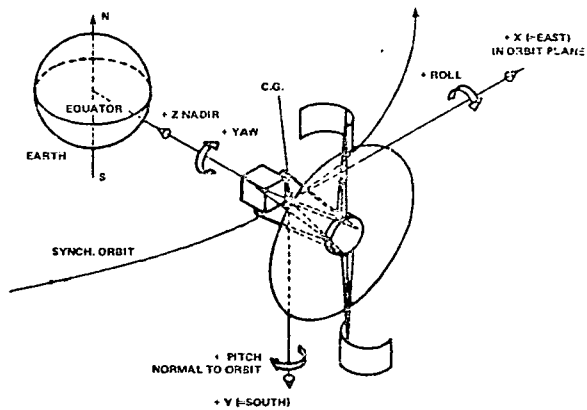


Figure V-3. Spacecraft Reference Orientation

1. Local Vertical Hold
2. Reference Orientation
3. Offset Point
4. Station Point/Nulling
5. Low Altitude Satellite Tracking
6. Slewing Maneuver
7. Antenna Pattern
8. Low Jitter
9. Ground Control

All of these modes are capable of being activated or deactivated by command from the ground.

Local Vertical Hold—This operational mode aligns the Z-axis of the spacecraft with the local vertical to an accuracy of ±0.1 degree in pitch and roll and aligns the Y-Z plane with Polaris to an accuracy of ±0.1 degree in yaw.

Reference Orientation—This mode maintains the local vertical, orbit plane reference orientation. The reference orientation of the spacecraft is the +Z (yaw) axis directed toward the center of the Earth (aligned with the local vertical), the +X (roll) axis lying in the orbit plane nominally in the direction of motion, and the +Y (pitch) axis perpendicular to the orbit plane, forming a right-hand coordinate system. This mode is initiated by a command from the ground.

Offset Point—This mode is provided to offset point the +Z axis of the spacecraft to any angle within 10 degrees of local vertical while using the primary sensors. A command from the ground initiates this mode. Also provided is the capability to switch to standby or alternate sensors by command from the ground.

Station Point/Nulling—The Station Point/Nulling mode nulls the roll/pitch errors and points the +Z axis of the spacecraft to a specified ground station. This mode is initiated by command from the ground.

Low Altitude Satellite Tracking—The satellite tracking mode causes the +Z axis of the spacecraft to point at a low Earth-orbiting satellite up to off-vertical angles of 11 degrees. This mode is initiated by command from the ground. Commands governing the angle, generated on board the spacecraft, cause the +Z reference axis to point to the low orbiting satellite. These commands are computed on board ATS-F using orbital parameters of the spacecraft and of the satellite being tracked.

Slewing Maneuver—This operational mode implements slew maneuvers based on sequential roll/pitch angle commands and is initiated by command from the ground.

Antenna Pattern Maneuvers—This mode is a special clover leaf pattern of roll, pitch and diagonal roll/pitch slews that measures the antenna gain pattern. Commands generated sequentially on board the spacecraft cause the +Z axis to trace out four main arcs in the North-South, East-West, and two diagonal directions, all of which are centered about the spacecraft/ground station line of sight. This mode can be initiated from the Station Point Nulling mode using the monopulse as an input for roll and pitch.

Low Jitter—This operation mode, by achieving a stable low jitter, provides an Earth referenced platform for conducting special Earth scanning experiments. This mode is also initiated by command from the ground.

Ground Control—This mode controls the inertial wheels or jet torquers by ground command using telemetered sensor data. The sensor data flows from the spacecraft computer on the ground and the computer actually commands the wheels and torquers. Control of roll, pitch and yaw by closed loop via a ground station is accomplished by using the interferometer with two ground stations.

DESCRIPTION OF CONTROL SYSTEM

Introduction

A block diagram of the acquisition control modes is shown in Figure V-4. The sensors used are shown in their order of use. The inertial reference assembly (IRA) provides rate damping signals. The coarse and fine sun-sensor signals are used to acquire the sun. The earth sensor is used for earth acquisition and provides the desired local vertical reference (roll, pitch, for subsequent operational control modes. The Polaris sensor is used for Polaris acquisition and subsequently provides a reference for the control of the yaw axis. The VHF monopulse provides a backup capability for roll and pitch acquisition of the earth. A standby redundant Polaris sensor provides for a yaw backup. The acquisition modes use the redundant propulsion system for torquing the spacecraft. Backup control modes at reduced power levels include an analog backup controller (ABC) and ground command of the spacecraft propulsion system (SPS) jets based on telemetered sensor data.

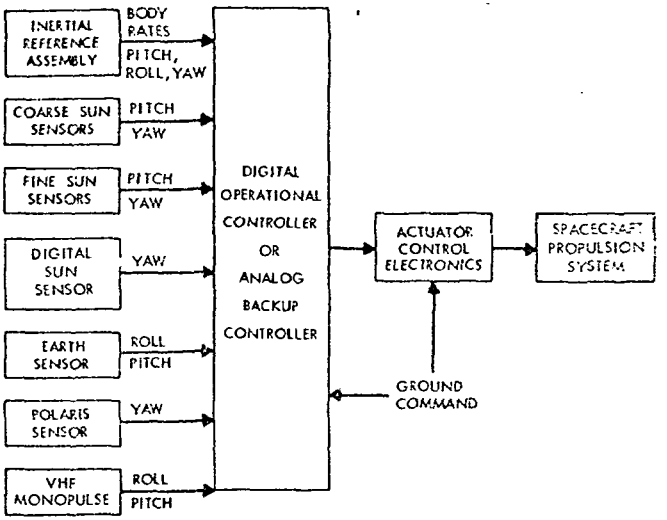


Figure V-4. Acquisition Control Modes Block Diagram

The block diagram of Figure V-5 shows the primary operational control modes and the mode in which each of the sensors may be used. Only one of two digital operational controllers is shown; however, a duplicate set of interfaces is provided so that either controller may perform the functions. The inertia wheels are the prime means for torquing the spacecraft with the propulsion system providing torques for unloading the wheels. The SPS is also capable of performing all modes except low jitter without the wheels. For orbit control, the SPS is the prime means of producing torque.

Table V-1 summarizes the operational backup modes of control.

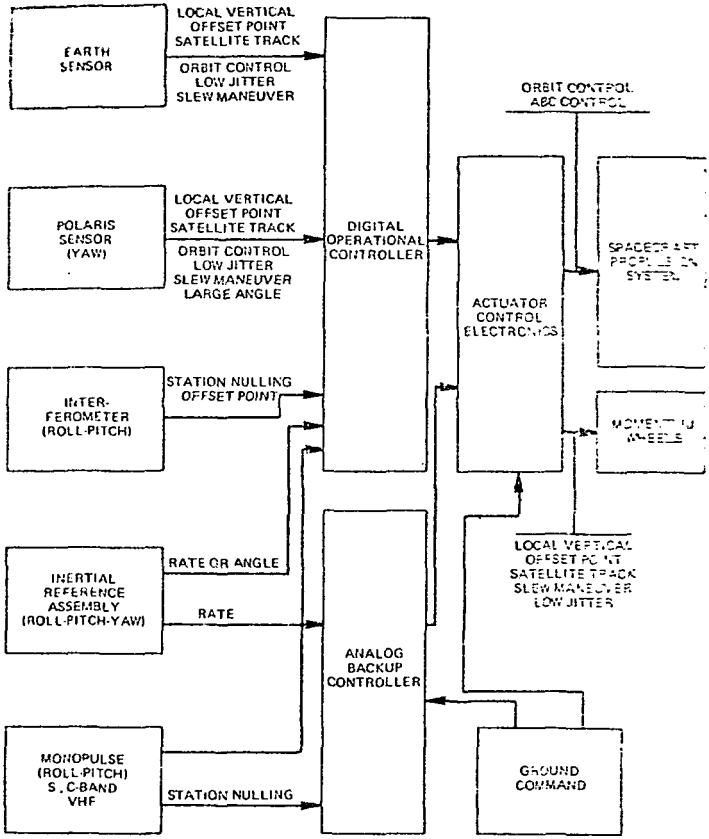


Figure V-5. Operation Control Modes Block Diagram

Table V-1

Backup Capability-Operational

Sensors		Controllers	Torquers
Roll/Pitch	Yaw		
Earth sensor	Polaris sensor 1	Digital operational controller 1	Wheels-SPS 1
Interferometer	Polaris sensor 2	Digital operational controller 2	Wheels-SPS 2
Monopulse	Yaw gyro (IRA)		SPS 1
(C-, S-band, VHF)			SPS 2
Earth sensor	Polaris sensor 1	Analog backup controller	SPS 1
Monopulse	Polaris sensor 2		SPS 2
(C-, S-band, VHF)			
Earth sensor	Polaris sensor 1	Ground computation and command	Wheels-SPS 1
Interferometer	Polaris sensor 2		Wheels-SPS 2
Monopulse	Interferometer		SPS 1
(C-, S-band, VHF)			SPS 2

Sensors

The sensors outlined above are described in the following paragraphs.

Inertial Reference Assembly—The three-axis inertial reference assembly is used during the acquisition mode and acquisition of Polaris. It can provide some temporary attitude hold functions and is the primary 3-axis sensor for the transfer orbit failure mode. The IRA consists of three gas bearing spinmotor gyros, with associated electronics, that provide output pulses corresponding to incremental body angle changes about each of the three axes. By counting the pulses, the total attitude can be calculated. The DOC also uses these pulses in its computations to provide body rate signals for rate damping and acquisition modes. The IRA also provides analog rate output signals for the interface with the ABC. Filtering of the signals is accomplished in the ABC.

Coarse-Null Sun Sensors—Four coarse-null sun sensors are provided, each with two-axis sensitivity. Each sensor makes use of silicon photovoltaic diode eyes. Each eye provides an analog output signal related to the angle between the sun line and a vector normally incident on the sensor. When used in conjunction with the electronics, the output from two differentially connected eyes provides a null when the sun is at equal angles relative to each of them.

The function of the coarse-null sun sensors is to enable the spacecraft X-axis to be brought to within 8 degrees of the sun line, where the fine sun sensor can take over control of the spacecraft.

Fine-Null Sun Sensors—Two fine-null sun sensors are provided, each with two-axis sensitivity. Each sensor consists of four silicon photovoltaic diode eyes interconnected to obtain a null output when the sensors are pointed to the sun. In addition, a target eye is included to provide an indication to the controllers for switching control from the coarse sensors to the fine sensors.

The function of the fine-null sun sensors is to bring the spacecraft X-axis to the sun line.

Digital Sun Sensors—The two digital sun sensors are used to update the IRA yaw gyro when the gyro is used to back up the Polaris sensor.

Earth Sensor—The earth sensor is used for earth acquisition during the acquisition modes and serves as the primary roll/pitch sensor for all operational modes. The earth sensor assembly consists of two sensing heads (one roll, one pitch) plus an electronics unit.

Polaris Sensor—The polaris sensor is used to acquire Polaris. It is the primary yaw sensor for all operational and most experimental modes. Two Polaris sensors are provided; however, only one is required for operation. Either sensor may be selected by command from the ground.

Monopulse—The monopulse, operating in S-band, C-band and VHF, provides pitch and roll acquisition of the earth by providing station nulling signals. The VHF monopulse is used as a backup capability.

Interferometer—When illuminated by RF energy from a ground transmitter, the interferometer measures the direction cosines between the spacecraft roll and pitch axes and the line-of-sight vector from the spacecraft to the transmitter. Digital outputs representing these two direction cosines are provided to the C&D and to telemetry covering the +17.5-degree roll pitch angle range of the interferometer. When two or more ground transmitters are

used, all three spacecraft attitude angles can be computed. The latter mode is implemented via telemetry and ground computation only.

Controllers

A ground control mode is provided using telemetered sensor data and control of the momentum wheels or jet torquers by command. Closed-loop control of roll, pitch and yaw via a ground station is possible using the interferometer with two ground stations.

The principle controllers are the digital operational controllers and the analog backup controller.

Digital Operational Controller—Two digital operational controllers are provided in the attitude control subsystem. The ground will choose the controller to be used. The digital operational controller serves as the prime controller for all modes of operation. In addition to performing control loop computations for stabilizing the spacecraft, the DOC provides the following functions:

- Accepts a wide variety of sensor inputs, solves control laws, and commands torquers for all modes
- Executes acquisition logic to acquire the sun, earth, and Polaris
- Accepts commands for offset pointing (angle and ground coordinate commands)
- Computes compensations required for pointing commands to account for orbit eccentricity, orbit inclination, and Polaris diurnal motion
- Computes commands for performing Z-axis tracking of a low altitude satellite based on transmitted ATS and satellite ephemeris data
- Computes commands for performing slew maneuvers for antenna pattern measurements
- Provides momentum wheel unload signals
- Provides a survival control mode if marooned in transfer orbit
- Provides a self-test routine within the DOC.

The basic DOC consists of a power supply, central processor, a memory, and an input/output section. The DOC provides for block transfer of data to the telemetry data acquisition and control unit or complete data transfer from memory on command from the ground.

The heart of the DOC is the central processor which performs programs stored within the memory. The input/output electronics accepts both analog and digital signals and can supply monitoring signals to telemetry, and control signals to sensors and to the ACE for controlling the momentum wheels and torquing jets.

Analog Backup Controller—The analog backup controller serves as a backup controller to the DOC in certain modes at reduced power levels. The ABC provides the following functions:

- Stabilizes the spacecraft
- Acquires the sun, earth, and Polaris with aid from the ground
- Maintains local vertical hold mode (0.5 degree accuracy)
- Maintains a station pointing mode (0.1 degree accuracy) with C-band monopulse.

The ABC has no digital interfaces. The basic control loop will consist of switching amplifiers with lag feedback compensation to obtain rate information (when not using gyros) for stabilizing the loop. The SPS jets, only, are used for attitude torquing.

Actuator Control Electronics (ACE)—The actuator control electronics will be energized at all times for all modes of operation. The ACE serves as the power amplifier between the controllers and the inertial wheels or the SPS, and provides outputs to 12 attitude control thrusters and 4 orbit control thrusters. The wheel drive section has outputs to three inertial wheels. The ACE also processes the wheel tachometer signals for telemetry to ground.

Torquers

The SPS and the momentum wheels provide thrust to torque the spacecraft for all modes and for attitude control. Each of the torquers are described in the following paragraphs.

Spacecraft Propulsion System—The SPS provides thrusts of suitable magnitude and direction to torque the spacecraft about the three axes for attitude control. The SPS also provides thrusts to impart changes in orbital velocity of the spacecraft for east-west and north-south orbit control.

Momentum Wheels—The momentum wheels serve as the prime torquers for all modes except acquisition, orbit control, and SAPPSAC. These wheels form a part of the ACS (one roll, one pitch, and one yaw). A pulse-type tachometer is provided for use by the DOC to obtain spacecraft rate information for the control loop. The tachometer output is also used to indicate the need for momentum wheel unloading.

SECTION VI

GROUND TERMINALS

Rosman Ground Station

Mojave Ground Station

Transportable Ground Station

Mobile Terminal

SECTION VI

GROUND TERMINALS

INTRODUCTION

Four ATS ground stations provide the main support to ATS operations for command and control of the ATS-F and -G spacecraft; for collection of range and range-rate data for orbit determination; for polarization data for spacecraft attitude determination; and for the performance of technological and scientific experiments. One station is located at Rosman, North Carolina and two are colocated at Mojave, California, one of which is the transportable. The Transportable Ground Station (TGS) will be moved overseas after the launch of ATS-F. The fourth, a mobile station, will provide an RF link for remotely locating the experimenters' unique equipment and will be used as backup support for the TGS. In addition, this terminal will contain UHF equipment to support the 860-MHz downlink for the ITV experiment. The mobile terminal will use a 21-foot antenna for C-band and UHF receive, and a 15-foot antenna for S- and L- bands.

The Rosman data-acquisition facility (with two 85-foot antennas) will be the primary ground station. However, the two secondary stations (each having a 40-foot and 15-foot antenna) will have the same basic capabilities as the primary station with regard to performance and evaluation of spacecraft data. The three stations, Rosman, Mojave, and the TGS will have nearly identical communications, and telemetry and command equipment. The Mojave and TGS facilities will each use the 40-foot antenna for C-band and the 15-foot antenna for S- and L- bands.

A separate transportable millimeter wave terminal for the Millimeter Wave Experiment will be located adjacent to the ATS facility at Rosman.

A more detailed description of these stations and their operation will be given later in this section.

All spacecraft maneuvers and experiments are controlled by direction from the ATS Operations Control Center (ATSOCC) at Goddard Space Flight Center (GSFC) at Greenbelt, Maryland. All test data obtained is forwarded from the ground stations to GSFC for processing, evaluation, and distribution to experimenters.

Rosman Ground Station

The Rosman Ground Station (Figure VI-1) is located in western North Carolina near the town of Brevard.

The Rosman II 85-foot antenna is used by ATS-F and -G spacecraft for C-band transmission and reception. This high-gain antenna with its feed and low-noise receiver systems provides a relatively quick conversion from STADAN to ATS communications use, and vice versa. This allows the antenna to be shared with other projects. Monopulse tracking techniques are used to automatically track the spacecraft. Polarization tracking at C-band is also provided. Communication signals are transmitted, received, and processed as required by the various ATS experiments. In addition, a Communications Experiment Test and Evaluation Console (CETEC) is used for evaluation of the spacecraft communications link.

A master control console (MCC) provides a focal point for ground station coordination. At the MCC, the mission director receives status indications and information that enable him to direct operations by issuing voice commands to the subsystem operators. Communications with ATSOCC is shown in Table VI-1.

Table VI-1

Ground Communications To and From Sites

<table>
<tr><td>Rosman</td><td>Mojave</td><td>TGS</td><td>Mobile</td></tr>
<tr><td colspan="4">Dial-Up Commercial Lines</td></tr>
<tr><td colspan="3">Two 7.2 KHz Data Links *</td><td>SCAMA</td></tr>
<tr><td>One 1-MHz Video Link, One-way Rosman to GSFC</td><td colspan="3"></td></tr>
<tr><td colspan="4">1 Full Duplex TTY Circuit #</td></tr>
</table>

* Includes SCAMA 1 circuit

2 FDX circuits at Rosman and Mojave from T-30 days, T+30 days

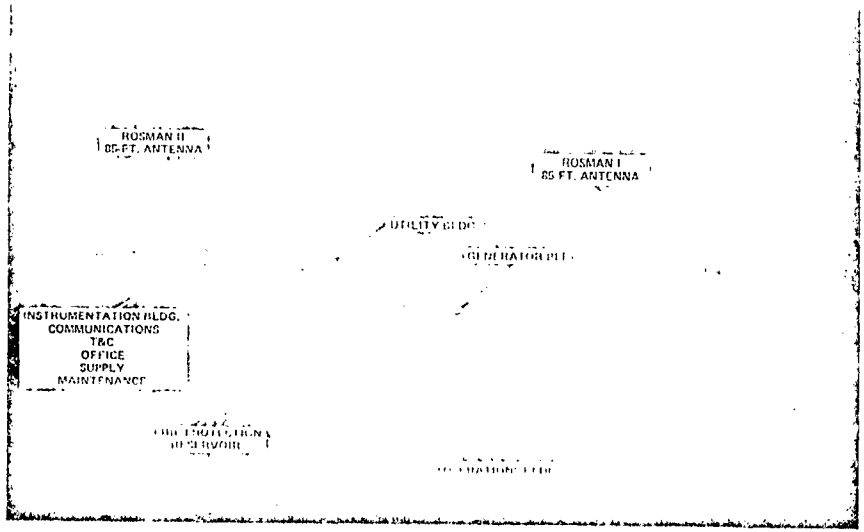


Figure VI-1. Rosman Ground Station

Spacecraft commands are encoded, transmitted and verified by the T&C system. Spacecraft housekeeping telemetry and discrete real-time functions are also received, processed, recorded and displayed by the telemetry subsystem.

All electric power at the station is provided by 10 diesel engine generators that supply 120/208-v, 60-Hz, 3-phase power. Six of these generators, rated at 250 kVA, are driven by Caterpillar D379 diesel engines. The other four, rated at 500 kVA, are driven by Caterpillar D398 diesel engines.

Table VI-2 lists the Rosman and Mojave station communications equipment characteristics and Table VI-3 presents the Telemetry and Command (T&C) subsystem functional characteristics for Rosman, Mojave and the TDS. Figure VI-2 is a functional block diagram of the T&C Subsystem for both Rosman and Mojave. A station block diagram for Rosman is shown in Figure VI-3.

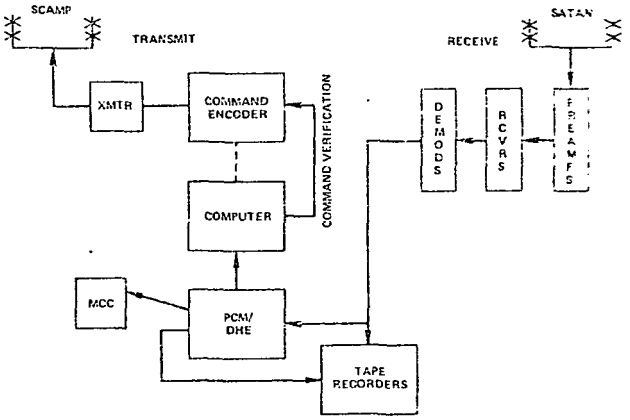


Figure VI-2. T & C Block Diagram for Rosman and Mojave

Table VI-2. ATS Ground Station Capabilities Chart

ITEM / STATIONS / DESCRIPTION	ROSMAN	MOJAVE	TRANSPORTABLE	MOBILE TERMINAL
LOCATION: Area	Near Ashville, N.C.	Goldstone Dry Lake, Mojave Desert, near Barrtow, Calif.	TBD	
Latitude	35 11'55" North	35 19'54" North	TBD	
Longitude	82 52'22" West	116 53'57" West	TBD	
ANTENNA C-BAND Type	Parabolic Reflector with STADAN Prime Focus Feed and ATS Cassegrain C-Band Feed	Parabolic Reflector with Cassegrain Feed		Parabolic Reflector with Prime Focus Feed
Diameter	85 ft	40 ft		21 ft
Polarization	Linear			
Mount	X-Y		AZ-EL	Polar
Receive Freq.	3.7 to 4.2 GHz			
Transmit Freq.	5.925 to 6.425 HGz			
*G/T (3.7 GHz) (Zenith)	38.4 db	31.5 db	32.0 db	24.8 db (Cooled)
Gain-Receive (3.7 GHz)	56.0 db	49.5 db	50.0 db	41.8 db
Gain-Transmit (5.9 GHz)	61.0 db	53.0 db	53.5 db	48.9 db
Receive System Noise Temp. (Zenith)	58°K	87°K		100°K Cooled 625°K Uncooled

*Minimum at peak unless otherwise noted.

Table VI-2. ATS Ground Station Capabilities Chart (Cont)

ITEM / STATION / DESCRIPTION	ROSMAN	MOJAVE	TRANSPORTABLE	MOBILE TERMINAL
ANTENNA (Cont)				
S- AND L-BAND				
Type	Parabolic Reflector with Prime Focus Feed			
Diameter	**15 ft			
Polarization	Circular			
Mount	AZ-EL			Polar
Receive Freq.	S-band: 2.05 to 2.1 GHz L-band: 1.5 to 1.58 GHz			
Transmit Freq.	S-band: 2.2 to 2.3 GHz L-band: 1.62 to 1.7 GHz			
G/T: L-Band (Zenith)	10.6 db			
G/T: S-Band (Zenith)	12.1 db			
Gain-				
S-Band RCV	35.7 db			
L-Band RCV	33.6 db			
S-Band XMT	36.9 db			
L-Band XMT	34.3 db			
Receive System Noise Temperature (Zenith)				
L-Band	200°K			
S-Band	230°K			
UHF				
Type				Parabolic Reflector with Prime Focus Feed

* The existing 40 ft L-band antenna at Mojave is being modified to include S-band capability.

Table VI-2. ATS Ground Station Capabilities Chart (Cont)

ITEM / STATION DESCRIPTION	ROSMAN	MOJAVE	TRANSPORTABLE	MOBILE TERMINAL
ANTENNA (Cont)				
UHF (Cont)				
Diameter				21 ft
Polarization				Circular
Mount				Polar
Receive Freq.				835-885 Mhz
G/T (Zenith)				5.2 db
Gain-Receive				31.7 db
Receive System Noise Temp. (Zenith)				450°K
MMW EXPRMNT				
Type	Parabolic			
Diameter	15 ft			
Feeds	Dual Frequency-Linear			
Mount	AZ-EL			
Receive Only Frequency	20 and 30 Ghz			
*G/T: 20 Ghz	28.0 db			
30 Ghz	29.5 db			
Gain: 20 Ghz	56 db			
30 Ghz	58 db			
Receive System Noise Temperature (Zenith)	20 Ghz - 630°K 30 Ghz - 710°K			

*Minimum at peak unless otherwise noted.

Table VI-2. ATS Ground Station Capabilities Chart (Cont)

ITEM / STATION / DESCRIPTION	ROSMAN	MOJAVE	TRANSPORTABLE	MOBILE TERMINAL
SPACECRAFT-GROUND COMMUNICATIONS SYSTEM				
SIMULTANEOUS CAPABILITY				
C-Band	Capable of transmitting one S/C channel and receiving two S/C channels simultaneously			Transmits one channel and receives one channel
C- and S-Bands	Receive and Transmit			
C- and L-Bands	Receive and Transmit			
VHF ANTENNA				
Receive -	SATAN		TACO	
Transmit -	SCAMP			
Gain:				
Receive	22 db		20 db	
Transmit	17 db			
Receive Beamwidth (P/2)	13			
Transmit Beamwidth (P/2)	25			
RECEIVER (FM)				
C-BAND				
Noise Temp	180°K			
RF Bandwidth	500 MHz			
IF Bandwidth	Up to 50 MHz			

Table VI-2. ATS Ground Station Capabilities Chart (Cont)

ITEM / STATION DESCRIPTION	ROSMAN	MOJAVE	TRANSPORTABLE	MOBILE TERMINAL
SPACECRAFT-GROUND COMMUNICATIONS SYSTEM (Cont)				
RECEIVERS (FM) (Cont)				
S- AND L-BAND				
Noise Temp	100°K			
RF Bandwidth	Up to 50 MHz			
IF Bandwidth	Up to 50 MHz			
TRANSMITTER (FM)				
C-BAND				
Max. Power Output	8 kW			2 kW
RF Bandwidth	50 MHz			
S- AND L-BAND				
Max. Power Output	S-Band: 100 W (CW) L-Band: 1 kW (CW)			
RF Bandwidth	S-Band: 50 MHz L-Band: 10 MHz			
GROUND COMMUNICATIONS TO AND FROM SITE	1 SCAMA Voice Circuit, 2 Full Duplex TTY Circuits			
	One 6-MHz Color TV link with 6-kHz audio link, one-way Reunna to GSFC	Dial-Up Commercial Line		
	One 1-MHz Data Link			
	Dial-Up Commercial Line			

Table VI-3

T & C Functional Characteristics for Rosman, Mojave, and TGS

Item Description		Rosman and Mojave	Transportable
ANTENNA	TYPE	SATAN Transmit - 4.9 Element Crossed Yagis Linear or Circular Polarization Receive - 16.8 Element Crossed Yagis Linear or Circular Polarization	TACO Transmit - 1.9 Element Crossed Yagi Linear or Circular Polarization Receive - 9.8 Element Crossed Yagis Linear or Circular Polarization
	Mount	X - Y	AZ - EL
	Receive Gain	22.0 db	20.6 db
	Transmit Beamwidth (P/2)	20°	20°
	Receive Beamwidth (P/2)	13°	13°
Transmitter	Maximum Power Output	2.5 kw	2.5 kw
	RF Bandwidth	3 MHz	3 MHz
	Type of Modulation	FM or AM	FM or AM
	FM Deviation Range	20 kHz Max	20 kHz Max
Receiver	Type	Dual Channel with Pre-Detection Combining	Dual Channel with Pre-Detection Combining
	Noise Figure	3.5 db	3.5 db
	Types of Demods.	FM or AM	FM or AM
	IF Bandwidth	10 kHz - 3 MHz	10 kHz - 3 MHz
Multiplex Equipment	No. of Channels	3	3
	Output Bandwidth	300 Hz - 12 kHz	300 Hz - 12 kHz

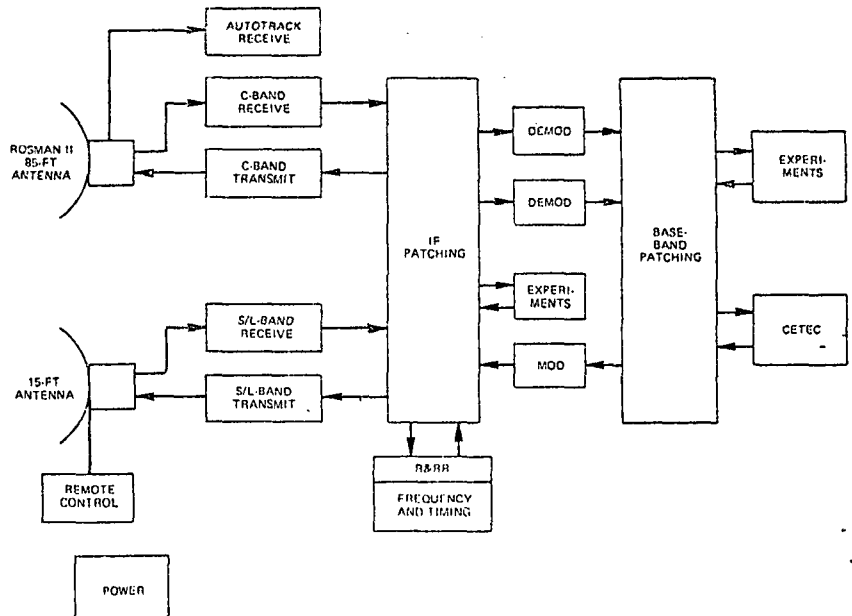


Figure VI-3. Rosman Ground Station Block Diagram

Mojave Ground Station

The Mojave Ground Station (Figure VI-4) is located in the Mojave Desert approximately 50 miles north of Barstow, California.

The Mojave facility uses the 40-foot parabolic antenna for C-band and the 15-foot antenna (not shown in the photograph) for S- and L-bands. The 40-foot antenna uses monopulse techniques to track the spacecraft at C-band. Polarization tracking is also provided. The C-, S-, and L-band communications system is the same as Rosman except for the size of the C-band antenna.

As in the Rosman station, an MCC is the focal point for coordination within the station. Status indications and information are received at the MCC that enables the mission director to direct operations by voice commands to the operators of the station subsystems. Communication with ATSOCC is accomplished by two SCAMA voice circuits and one full duplex TTY circuit. (See Table VI-1.)

Spacecraft commands are encoded, transmitted and verified by the T&C system. See Figure VI-3 for the T&C block diagram. Spacecraft housekeeping telemetry and discrete real-time functions are also received, processed, recorded, and displayed by the telemetry subsystem.

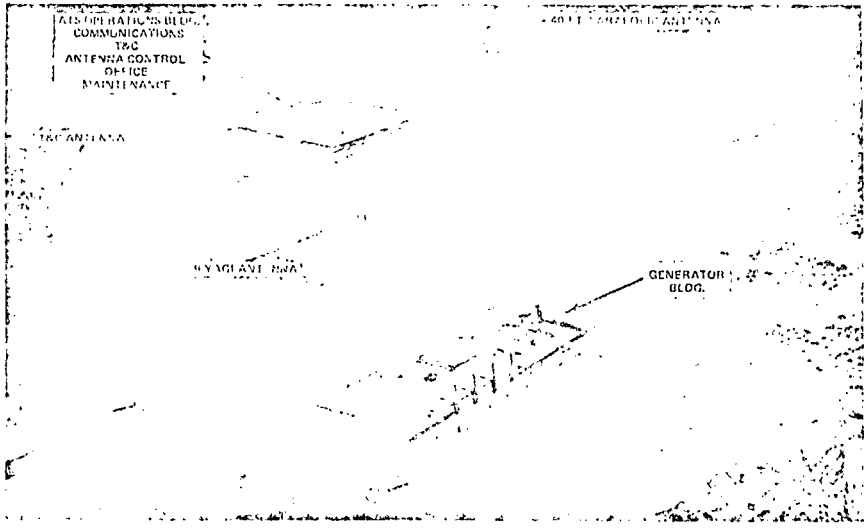


Figure VI-4. Mojave Ground Station

Station power is provided by 2500-kW gas turbine driven generators.

See Table VI-2 for the Mojave station communications equipment characteristics, and Table VI-3 for the T&C subsystem functional characteristics. Figure VI-5 shows the Mojave Ground Station Block Diagram.

Transportable Ground Station (TGS)

The Transportable Ground Station (Figure VI-6) is operationally quite similar to the Mojave station. The antenna is constructed in such a manner as to allow it to be dismantled, transported, and reerected in a remote area.

The station is composed of four 40-foot, air-conditioned trailers that house the electronics equipment. Three of these trailers have removable sides so that, when on site, they can be mated together side by side to form a single complete Operation Control Complex (OCC) with access to all subsystems, with the exception of T&C. The T&C system is contained in a separate trailer. This system, which is basically the same for all of the ground stations, is integrated with the OCC so that overall control of the station can be maintained from one point. Spacecraft commands are encoded, transmitted and verified by the T&C system. Spacecraft housekeeping telemetry and discrete real-time functions are also received, processed, recorded and displayed by the telemetry subsystem.

One advantage of this type of transportable ground station is that the T&C system can be physically separated from, and operated independently of, the other vans of the ground station as long as power is provided to it.

The station master control console, from which all monitoring and control functions are handled, is located in the center trailer. Communications with ATSOCC is accomplished by one SCAMA voice circuit and two full duplex TTY circuits.

Within the area of the OCC complex is the supporting equipment, including three office vans, a maintenance van, two storage vans, and a communications van. Power and signal cables are routed through shallow covered trenches to interconnect the various elements of the station.

The 40-foot antenna, with its Cassegrain type feed system, uses precision monopulse techniques to track the spacecraft. Polarization tracking is also provided. The C-, S-, and L-band communications system is the same as Mojave.

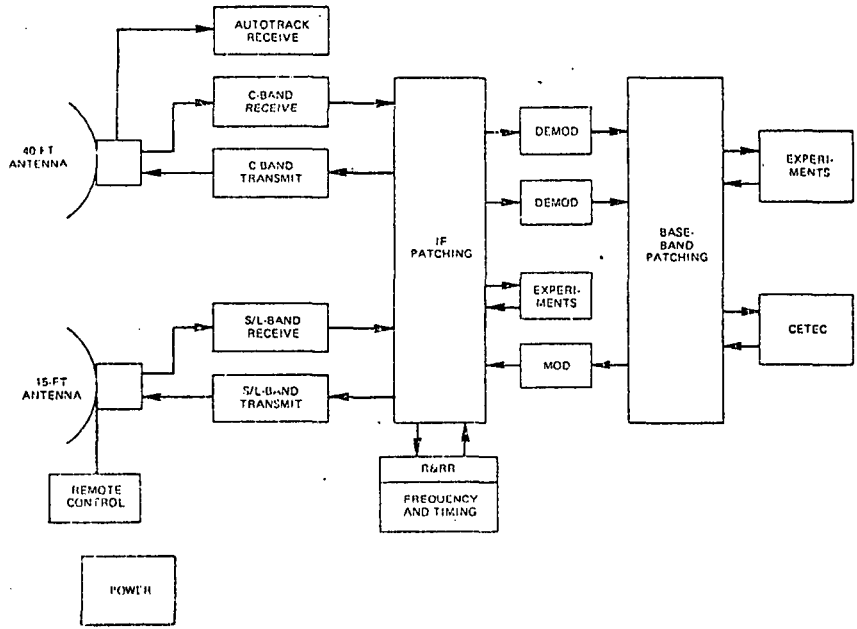


Figure VI-6. Mojave Ground Station Block Diagram

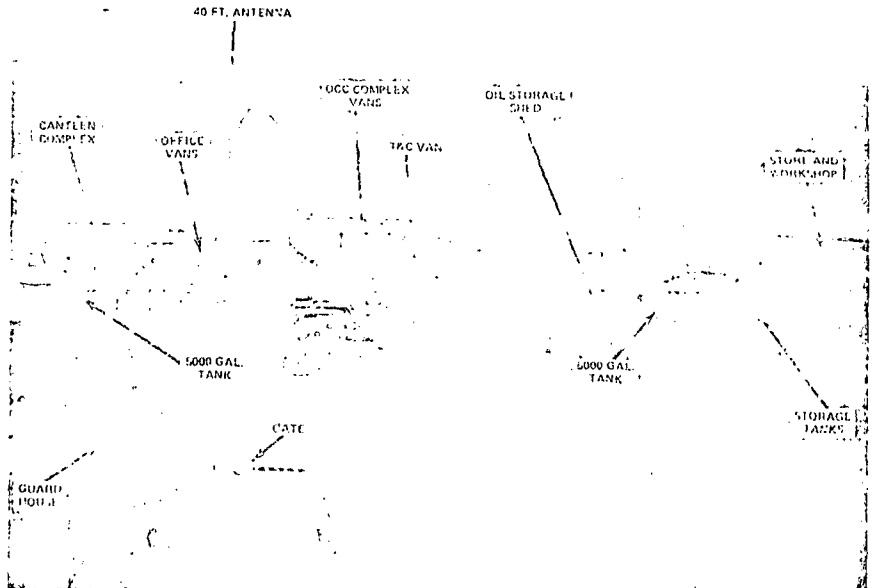


Figure VI-8 Typical Transportable Ground Station

The T&C functions are transmitted over a SCAMP antenna and received via a TACO antenna. Since the beamwidth is broad for these antennas, they do not need periodic alignment with the synchronous spacecraft.

Station power is provided by Diesel driven generators with a total capacity of about 1000 kVA. Backup power is produced by two diesel-driven generators producing 624 kVA.

The T&C block diagram for this station is presented in Figure VI-7. Table IV-1 lists the communications equipment characteristics for the TGS and Figure VI-8 shows the TGS block diagram.

Mobile Terminal

The ATS-F and -G Mobile Terminal, shown in Figure VI-9, will be located at a remote site. It is capable of transportation over normal highways at maximum legal speeds and at low speeds over improved roads to the installation sites. The terminal can be shipped by air or sea. Figure VI-10 shows the vans in the transport configuration.

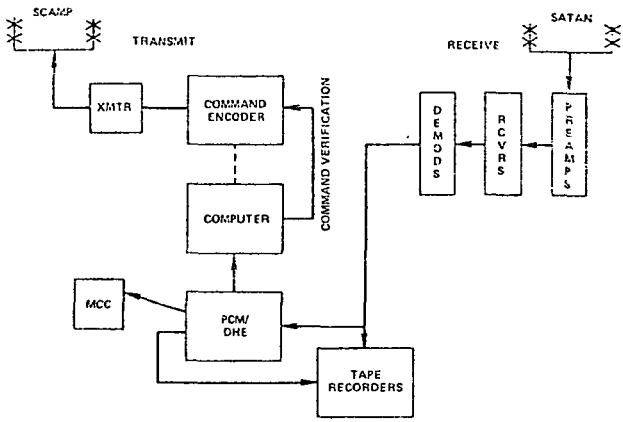


Figure VI-7. T & C Block Diagram for TGS

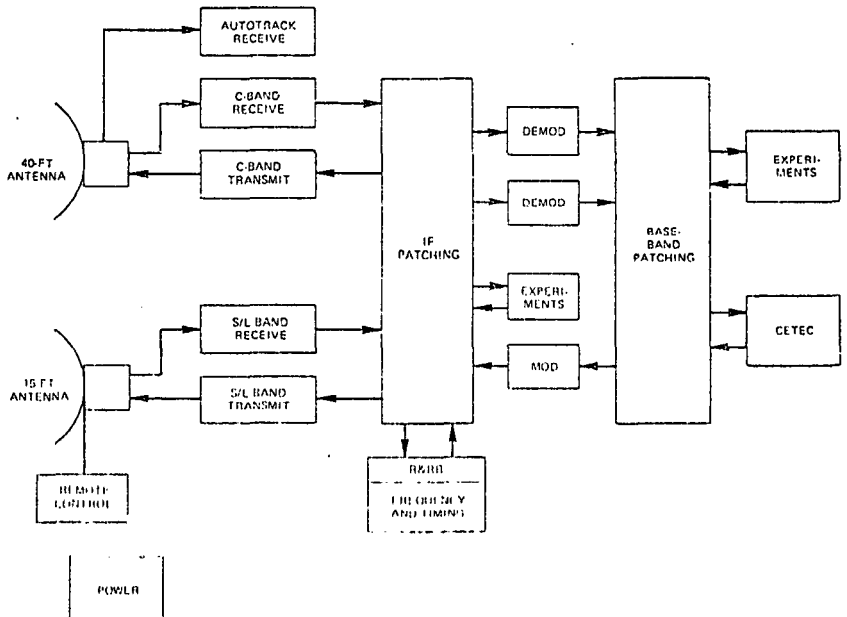


Figure VI-8. TGS Block Diagram

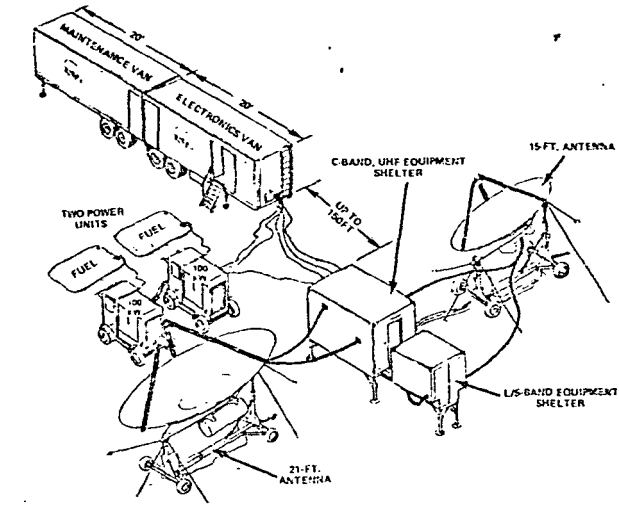


Figure VI-9. ATS-F and -G Mobile Terminal

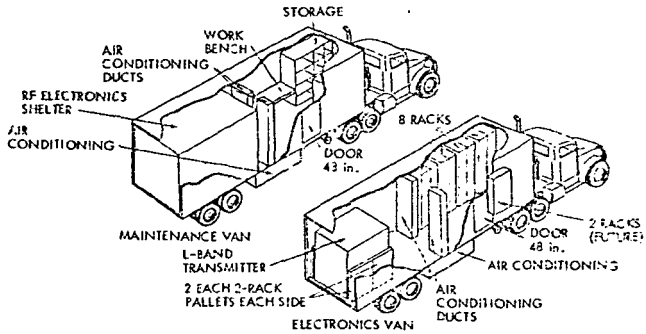


Figure VI-10. Vans in Transport Configuration

The mobile terminal is equipped to transmit and receive simultaneously on two bands, as shown below.

Operational Bands	Function
C- and S-band	Receive and transmit
C- and L-band	Receive and transmit
C-band and UHF	Receive and transmit on C-band but receive only on UHF

The antenna subsystem is capable of operating with geostationary synchronous satellites. In addition to transmission and reception of communications, the antenna subsystem is also used for range and range-rate signals.

The terminal is required primarily for conduct of the following:

- Television Relay Using Small Terminals (TRUST) experiment
- Position Location and Aircraft Communications Experiment (PLACE)
- Ranging
- Backup support to the Transportable Ground Station during maintenance, modifications, and overhaul of communications subsystems
- Emergency support for communications in remote locations
- Demonstrations
- To accommodate experiments on ATS-G.

To accomplish the above, the subsystems, described below, constitute the mobile terminal.

- 21-foot polar mount antenna with C-band transmit and receive, and UHF receive focal point feeds

- 15-foot polar mount antenna with S- and L-band transmit and receive focal point feeds
- 70-MHz wideband modulator
- 70-MHz wideband demodulator
- C-, S-, L-band transmitters including upconverters, drivers, and power amplifiers
- Test translator and performance monitors
- C-, S-, L-band and UHF receivers including low noise pre-amplifiers, post amplifiers and wideband modulators
- Power generating and distributing equipment
- Miscellaneous equipment including racks, pallets, intercommunications system, IF patching, baseband patching, and time standardization equipment

Station power is provided by two diesel engine generators as part of a power subsystem. Each generator is 3-phase, 4-wire type and has an output of 100 kW at 60 Hz, 120/208 vac.

Communication with ATSOCC is accomplished by one SCAMA voice circuit and one full duplex TTY circuit.

Table VI-4 lists the transmitter and receiver characteristics, and a station block diagram is shown in Figure VI-11. System parameters are shown in Table VI-5, and Tables VI-6 and VI-7, show parameters for the 15-foot and 21-foot antennas, respectively.

Table VI-4
RF Characteristics for the Mobile Terminal

Frequency Bands of Operation	Function	Antenna Gain (db)	Polarization	Transmit Power	System Noise Temperature (°K)
C-Band		21 ft.			
5925 to 6425 MHz	Transmit	48.9	Rotatable Linear	2 kW	—
3700 to 4200 MHz	Receive	44.8	Rotatable Linear	N/A	100 (cooled mode [illegible] (warm, second [illegible]
S-Band		15 ft.			
2200 to 2300 MHz	Transmit	36.9	LCP/RCP	100 W	—
2050 to 2100 MHz	Receive	35.7	LCP/RCP	N/A	250
L-Band		15 ft.			
1620 to 1700 MHz	Transmit	34.3	LCP/RCP	1 kW	—
1500 to 1580 MHz	Receive	33.6	LCP/RCP	N/A	250
UHF-Band		21 ft.			
635-885 MHz	Receive only	31.7	LCP/RCP	N/A	450

Note: All gain is based on a 50% efficiency across the bands.

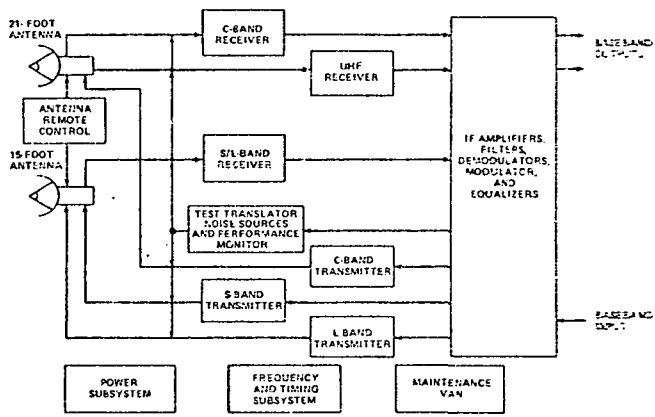


Figure VI-11. ATS-F and -G Mobile Terminal System Block Diagram

Table VI-5
Mobile Terminal System Parameters

	C-Band	S-Band	L-Band	UHF Band
Transmit Frequencies (GHz)	5.925 to 6.425	2.2 to 2.3	1.62 to 1.7	
Receive Frequencies (GHz)	3.7 to 4.2	2.05 to 2.10	1.5 to 1.58	0.835 to 0.[illegible]
Power Amplifier Maximum Output (dBW)	33.0	20	30	
EIRP (dBW)	79.9	54.9	62.3	
System Noise at Zenith (°K)	100 cooled 625 uncooled	230.0	200	400

Table VI-6
15-Foot Antenna Subsystem Parameters

Parameter	Value
Gain at 2200 to 2300 MHz 2050 to 2100 MHz 1620 to 1700 MHz 1500 to 1580 MHz	36.9 db 35.7 db 34.3 db 33.6 db
Antenna Diameter	15 feet
Feed Type	Focal point
F/D Ratio	0.42
Polarization	LCP/RCP manually switchable
Mount Type	Adjustable hour angle declination
Angular Coverage	
Hour Angle Polar Angle Declination	± 90 degrees 0 to 60 degrees ± 7 degrees
Velocity Capability	0.2 deg/sec each axis in 70-mph winds
Acceleration Capability	0.2 deg/sec^2 each axis nominal
Pointing Accuracy	0.01 deg/sec^2 in 70-mph wind < 0.128 deg in 50-mph wind (3σ)
Control Mode	Open loop control of AC drive motors
Backup Control Mode	Hand cranks on each axis
Angular Display	Analog dials with 1:1 and 36:1 Synchros for reading out angles

Table VI-7
21-Foot Antenna Subsystem Parameters

Parameter	Value
Gain at 5925 to 6425 MHz 3700 to 4200 MHz 835 to 885 MHz	48.9 db 44.8 db 31.7 db
Antenna Diameter	21 feet
Feed Type	Focal point
F/D Ratio	0.4
C-Band Polarization	Rotatable, linear, motor driven
UHF Polarization	LCP/RCP
Mount Type	Adjustable hour angle/declination
Angular Coverage Hour Angle Polar Angle Declination	 ±90 degrees 0 to 60 degrees ±7 degrees
Velocity Capability	0.2 deg/sec each axis in 70-mph winds
Acceleration Capability	0.2 deg/sec^2 each axis nominal 0.01 deg/sec^2 in 70-mph wind
Point Accuracy	<0.128 degree in 50-mph wind (3σ)
Control Mode	Open loop control of AC drive motors
Backup Control Mode	Hand cranks on each axis
Angular Display	Analog dials with 1:1 and 36:1 Synchros for reading out angles

SECTION VII

TECHNOLOGICAL EXPERIMENTS

Very High Resolution Radiometer Experiment

Position Location and Aircraft Communication Experiment

Instructional Television Experiment

Television Relay Using Small Terminals

Millimeter Wave Experiment

Tracking and Data Relay Experiment

Spacecraft Attitude Precision Pointing & Slewing Adaptive Control Experiment

Cesium Bombardment Ion Engine Experiment

ATS-F 13- and 18-GHz Propagation Experiment

Laser Retroreflector Experiment

Educational Television Experiment

SECTION VII

TECHNOLOGICAL EXPERIMENTS

VERY HIGH RESOLUTION RADIOMETER EXPERIMENT

This experiment is directed toward:

- The development of meteorological applications with registered visible and infrared measurements from a geosynchronous satellite
- Testing of a precision radiometer on a three axis stabilized spacecraft.

The radiometer provides both day and night measurements for the determination of cloud motions, tropical storm life cycles, extratropical storm life cycles, mesoscale phenomena, and cloud climatology studies. In addition, the radiometer system supplies information used for earth albedo measurements, earth resource and ocean temperature studies. It operates on two different frequency ranges, one in the infrared between 10.5-12.5 μm and one in the visible spectrum between 0.55-0.75 μm. With an 8-inch Cassegrainian telescope as the energy collecting optic, the instrument generates a picture in 24 minutes with a resolution of 5.5 and 11 km at the subsatellite point for the visible and infrared channels, respectively.

The radiometer data will be reduced and analyzed by a combination of digital computer and photo imaging techniques. In addition to recording all radiometer data on magnetic tape while the spacecraft is within view of the United States, the Principal Investigator will be provided with near real time (within 1 hour) readout of selected frames of radiometer data. Data will be recorded on magnetic tapes which will be mailed to Goddard on a regular basis.

Data may be recorded simultaneously on a separate magnetic tape or tapes which can be used by potential investigators. Formats of the data and other pertinent information will be provided by the Principal Investigator. It would be expected that anyone desiring to participate in this fashion would provide the necessary tape recorder and magnetic tapes.

POSITION LOCATION AND AIRCRAFT COMMUNICATION EXPERIMENT

The purpose of the position location and aircraft communication experiment (PLACE) is to obtain engineering data and practical experience for determining the operational feasibility of an air traffic control (ATC) satellite system operating in the aeronautical L-band.

The PLACE system is shown in Figure VII-1 and consists of the ATS-F parabolic antenna, the ATS-F communication transponder, SSB aircraft transmitters and receivers, and primary and alternate control centers.

There are two primary objectives of this experiment. The first is to prove the feasibility of two-way communications between ground terminals and aircraft. Included in this objective are:

- The use of ATS-F as a synchronous satellite for relaying all communications
- The use of the aeronautical L-band for satellite-to-aircraft links
- The use of both voice and digital two-way communication
- The use of multiple access through a satellite for both aircraft-to-ground and ground-to-aircraft links.

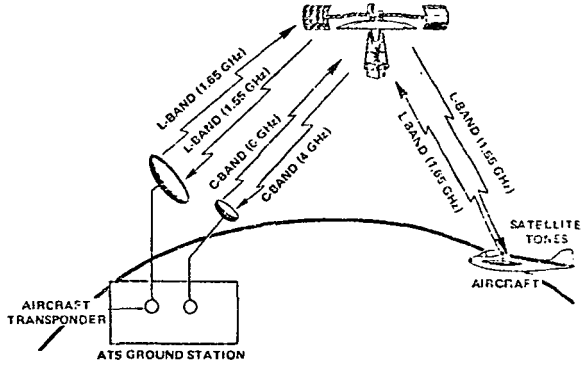


Figure VII-1. Position Location and Aircraft Communication Experiment System

PLACE permits as many as 100 airborne terminals to communicate with a ground terminal through the ATS-F spacecraft in a completely coherent system. The links are established by first phase locking the ATS-F communications transponder to a C-band signal from the ground station. The C-band signal is processed through the receiver and IF amplifier. It is then coherently upconverted to L-band at 1550 MHz and transmitted to the aircraft transponder which phase locks to the L-band signal. When this is done, all stations are coherent to each other. Information from the aircraft is transmitted in the form of single sideband modulated signals in 10-kHz channels in the L-band from 1650 MHz to 1651 MHz. ATS-F receives the spectrum from 1650 MHz to 1652 MHz and, after preamplification, downconverts it to baseband in the IF amplifier. This baseband signal is used to phase modulate a carrier (coherent to the C-band signal) which is then upconverted to C-band and is processed through the transmitter. The ground station receives the subcarrier and recovers the aircraft's signal.

The special multiple access capability required provides for simultaneous operation of C-band to L-band, and L-band to C-band. To achieve this, the communications subsystem provides a receiver linearity that prevents the multiple carriers from producing significant intermodulation components. When the multiple carriers are received, they are translated to baseband frequencies and then phase modulated onto a single carrier by the use of selectable baseband filters. The phase modulation index is held by automatic gain control at a constant 1 radian, independent of the number of carriers.

The second objective is to investigate the feasibility and to evaluate the absolute and relative accuracies of several position location techniques using a single satellite. Each of these techniques relay various signals from the aircraft via the satellite to the control center for data processing and position determination.

The above objectives will be achieved through the use of three types of experiments:

- Ground Based Engineering
- Ground Based Simulation
- In-Flight Performance

The ground based engineering experiments include system calibration needs, link performance measurements, multiple-access performance, power and frequency control techniques, system communication capacity, and quality and ranging precision for the various techniques being evaluated.

The ground based simulation experiments include the use of noise loading and simulated signals and the determination of the system performance parameters.

In addition, the position location and tracking accuracy performance for both fixed and mobile simulated aircraft terminals will be determined.

In-flight performance experiments will be conducted in close cooperation with the FAA and possibly other coexperimenters. These will include actual aircraft flights to determine the effects of multipath, ionosphere, noise environment, and geographic location on both the L-band communications and position location links. Operational experiments including communication link utilization, multiple aircraft tracking, capacity limitations, possible extensions, and ground and cockpit terminals will form an important part of the combined GSFC and FAA research effort.

Others could participate in the PLACE experiment by

- providing the special purpose ground equipment needed at the primary ground station,
- providing a complete secondary ground station,
- providing a terminal to simulate one or more aircraft, or
- providing aircraft terminal and actual aircraft flights to acquire data.

The Principal Investigator will act as the coordinator for all participants. Detail performance specifications can be obtained from the Principal Investigator by all potential participants.

Operating frequencies are given in Table VII-1.

INSTRUCTIONAL TELEVISION EXPERIMENT

This experiment which is known variously as the ITV (Instructional TV Experiment and SITE (Satellite Instructional Television Experiment), is a joint experiment between the NASA and the Department of Atomic Energy, Government of India, entered into formally on September 18, 1969.

Table VII-1
PLACE Frequencies (GHz)

BANDS	GROUND STATION	
	Receive	Transmit
C	3.70 to 4.20	5.90 to 6.40
S	2.05 to 2.10	2.20 to 2.30
L	1.475 to 1.580	1.62 to 1.70

NOTES

1. Each system will be able to down-convert any carrier input frequency at any 10-kHz increment in the receiver input frequency band to 70 MHz IF.
2. Signal inputs to each transmitter are at 70 MHz carrier frequency. Output powers of transmitters are controllable up to 8 kw for C-band, 1000 w for L-band, and 100 w for S-band.

The general objectives of the experiment are to:

- gain experience in the development, testing, and management of a satellite-based instructional television system, particularly in rural areas and to determine optimal system parameters
- demonstrate the potential value of satellite technology in the rapid development of effective mass communications in developing countries
- demonstrate the potential value of satellite broadcast TV in the practical instruction of village inhabitants
- stimulate national development in India, with important managerial, economic, technological and social implications.

Specific Objectives

Indian Instructional Objectives: Primary—

- contribute to family planning objectives
- improve agricultural practices
- contribute to national integration

Indian Instructional Objectives: Secondary—

- provide a system test of broadcast satellite TV for national development
- enhance capability in the design, manufacture, deployment, installation, operation, movement, and maintenance of village TV receivers
- gain experience in the design, manufacture, installation, operation and maintenance of broadcast and/or distribution facilities to the extent that these are used in the experiment
- gain an opportunity to determine optimum receiver density, distribution and scheduling, techniques of audience attraction and organization and to solve problems involved in developing, preparing, presenting and transmitting TV program material.

United States Technical Objectives—

- test the design and functioning of an efficient, medium power, wideband space-borne FM transmitter, operating in the 860 MHz band and gain experience on the utility of this space application.

The experiment will be conducted for a period of one year starting approximately one year after the launch of ATS-F. The ATS-F will be positioned at approximately 35° E Longitude. A frequency modulated TV carrier at 6 GHz will be transmitted to the ATS-F earth coverage antenna from one of three earth stations: Ahmedabad, Delhi, or Poona. The signal is processed and retransmitted at both 4 GHz and 860 MHz. The 4 GHz downlink will be used to monitor the system performance.

The 860 MHz downlink will be used to test the concept of a hybrid system involving both direct reception by low-cost augmented TV receivers as well as higher sensitivity earth stations for rebroadcast at VHF to conventional TV receivers. System design parameters are such that an equivalent TASO grade 1 picture will be received by the direct reception terminal while a CCIR recommended quality signal will be received by the rebroadcast earth station. About 3,000 direct reception and 2000 conventional sets will be located in 5000 villages. The direct reception terminals will be located in clusters of about 400 each in various parts of India while the conventional sets will be located in villages near existing or planned VHF TV transmitters.

The nominal system performance is summarized in Tables VII-2, -3, -4 while the beam coverage is shown in Figure VII-2.

Table VII-2
Site Uplink Calculations

		Ahmedabad
Frequency	MHz	5950
Antenna Diameter	ft.	46
Antenna Gain	db	55.6
Transmit Power	KW	5
Miscellaneous Losses	db	1.5
EIRP (Peak)	dbw	90.5
Path Loss	db	199.4(A)
Spacecraft Receiver G/T (Peak)	db/° K	-17
Boltzman's Constant	db/° K-Hz	-228.6
Receiver Carrier to Noise Density	db-Hz	102.7
Carrier to Total Noise in Spacecraft 40 MHz Bandwidth	db	26.7

(A) Spacecraft at 35° E Elevation Angle = 41° => 0.45 db Additional path loss

Table VII-3
Site Downlink Calculations

		Transport Gnd. Station	Ahmedabad	Re-Diffusion Terminal	Small Terminal
Frequency	MHz	3750	3750	860	860
Antenna Diameter	ft.		46	30	1[illegible]
Antenna Gain	db		51.4	33.6(A)	[illegible]
System Noise Temp.	°K		130	1000	1000
Receiver G/T	db/° K		30.3	3.6	-6 [illegible]
Path Loss	db		195.4(C)	183.0(D)	183.[illegible]
S/C EIRP	dbw		28(E)	48(F)	48 [illegible]
Boltzman's Constant	db/° K-Hz		-228.6	-228.6	-228.6
Received Carrier to Noise Density	db-Hz		91.5	97.2	97.6

Table VII-4
Site Performance Calculations

IF Bandwidth	MHz	30/15	30/15	30 [illegible]
Carrier to Total Noise	db	16.7/19.7	22.4/25.4	22.8 25.8
Peak Video Deviation	MHz	5.2	5.2	5.2
Video Baseband Bandwidth	MHz	5.0	5.0	5.0
FM Improvement Factor	db	9.9/6.9	9.9/6.9	9.9 6.9
CCIR Noise Weighting	db	16.3	16.3	16.3
Peak-Peak Conversion	db	6.0	6.0	6.0
De-emphasis Noise Reduction	db	0	0	0
Peak-to-Peak Video/ Weighted Noise	db	48.9	54.6	48.9

(A) Spacecraft inclination may be as much as ± 1° when it reaches India, which is about the 2 db beamwidth of the 30 ft. antenna.
(B) Allows for a 1 db degradation in G/T with time.
(C) Spacecraft at 35° E, elevation angle = 41° = > 0.45 db additional loss.
(D) Worst possible elevation angle from Calcutta = 25° = > 0.76 db additional PL.
(E) On axis gain
(F) 3 db FOV

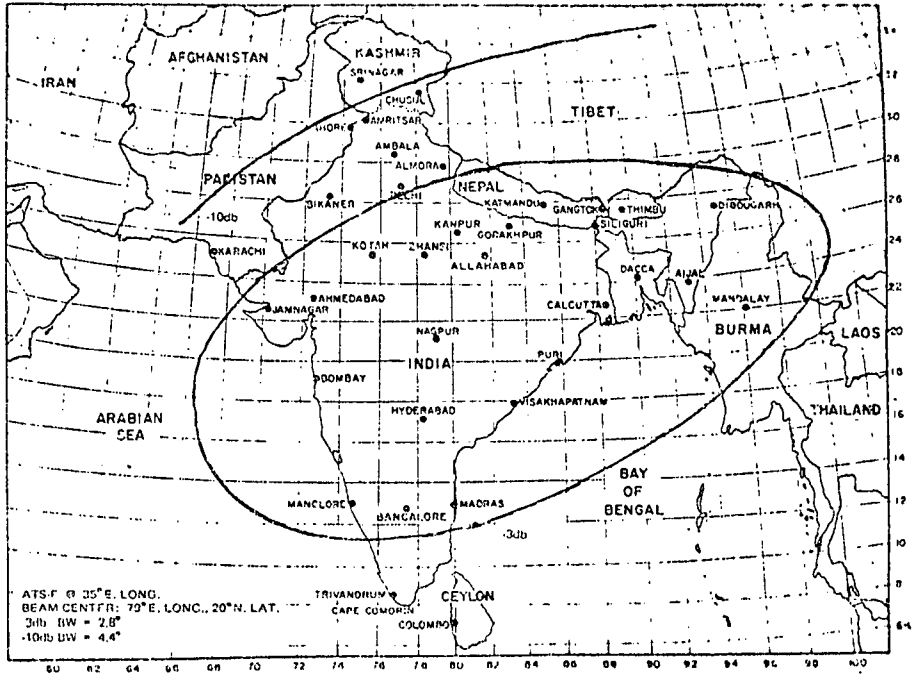


Figure VII-2. 860 MHz Antenna Beam Coverage of India for ITV

TELEVISION RELAY USING SMALL TERMINALS (TRUST)

The purpose of the TRUST experiment is to advance and promote the technology of wideband satellite communications to small ground terminals by developing and demonstrating a pilot system using the ATS-F spacecraft with its high-gain parabolic reflector. Specific program goals are:

- To test and evaluate an experimental system for FM relay of black and white and color television signals (and associated sound) between the ATS-F spacecraft and a UHF receiving facility
- To evaluate the performance of the pilot system relative to experiment design objectives and internationally recognized and accepted standards for television transmission systems
- To observe the effects of ionospheric dispersion on system performance as a function of electron density, ground station location, and other system variables, and compare with theoretical predictions
- To provide interested underdeveloped countries an opportunity to participate in tests and demonstrations of a high EIRP satellite suitable for national ETV using inexpensive receivers, and to provide advice and consultation in the design and implementation of suitable receivers for such systems.

The TRUST experiment basic system will consist of a high-power microwave transmitting terminal for earth-to-satellite communications, the ATS-F spacecraft with a microwave-to-UHF communications repeater, and a pilot mobile UHF ground receiving facility (see Figure VII-3). The earth-to-satellite communications will be accomplished by using existing NASA microwave facilities at Rosman Station, North Carolina, or Mojave, California, or both. Extensive research will be conducted to develop and evaluate low-cost, consumer quality receiving terminals. It is anticipated that terminal costs can be held to less than $200 in large volume production.

The RF links presented in Tables VII-5 and VII-6 can support wideband FM transmission of CCIR quality television signals (picture plus sound) and TASO Grade 1 with adequate threshold margin. Using parameters listed, the nominal system performance is summarized in Table VII-7.

An outline block diagram of the baseband and IF communication system configuration at Rosman and Mojave is shown in Figure VII-4

Table VII-5
TRUST Uplink Calculations

		Rosman	Mobile Terminal
Frequency	MHz	5950	5550
Antenna Diameter	ft	85	21
Antenna Gain	db	61.0	48.9
Transmit Power	KW	6	2
Miscellaneous Losses	db	2.0	2.0
EIRP	dbw	98.0	79.9
Path Loss	db	199.3(A)	200(B)
Spacecraft Receiver G/T (3 db FOV)	db/° K	-20	-20
Boltzman's Constant	db/° K-Hz	-228.6	-228.6
Received Carrier to Noise Density	db-Hz	107.3	98.5
Carrier to Total Noise in Spacecraft 40 MHz Bandwidth	db	31.3	12.5

(A) Spacecraft 94° W, Rosman Elevation Angle = 47° = > 0.35 db additional PL
(B) Mobile Terminal in NE Canada, Elevation Angle = 12° = > 1.05 db Additional PL

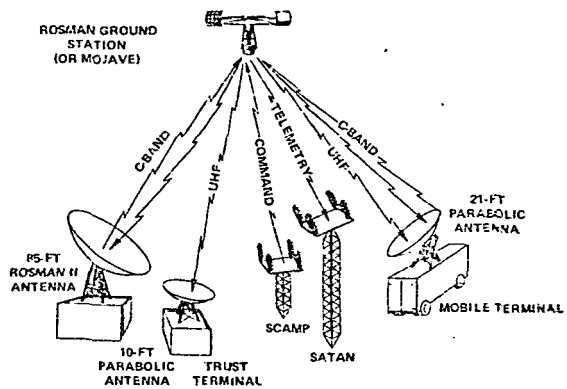


Figure VII-3. TRUST System

Table VII-6
TRUST Downlink Calculations

		Rosman	Mobile Terminal	Mobile Terminal	Small Terminal
Frequency	MHz	3750	3750	860	860
Antenna Diameter	ft.	85	21	21	10
Antenna Gain	db	56.0	44.8	31.7	25
System Noise Temp.	°K	58(A)	100(A)	450	1000
Receiver G/T	db/° K	38.4	24.8	5.2	-6(B)
Path Loss	db	195.3(C)	196.0(D)	183.3(D)	192.7(E)
Spacecraft EIRP (3 db FOV)	dbw	25	25	48	48
Boltzman's Constant	dbw/° K Hz	-228.6	-228.6	-228.6	-228.6
Receiver Carrier to Noise Density	db-Hz	96.7	82.4	98.5	87.9

Table VII-7
TRUST Performance Calculations

IF Bandwidth	MHz	15/30	15/30	15/30	15/30
Carrier to Total Noise Ratio	db	24.9/21/9	10.6/7.6	26.7/23.7	16.1/13.1
Peak Video Deviation	MHz	5.4	5.4	5.4	5.4
Video Baseband Bandwidth	MHz	4.0	4.0	4.0	4.0
FM Improvement Factor	db	10.2/13/2	10.2/13.2	10.2/13/2	10.2/13.2
CCIR Noise Weighting	db	10.2	10.2	10.2	10.2
Peak-to-Peak Conversion	db	6.0	6.0	6.0	6.0
De-Emphasis Noise Reduction	db	2.5	2.5	2.5	2.5
Peak-to-Peak Video/ Weighted Noise	db	53.8	39.5	55.6	45

(A) Antenna at zenith. T will increase slighly at lower elevation angles.
(B) Allows for a 1 db degradation in G/T with time.
(C) Spacecraft at 94° W, Elevation angle = 47° = > 0.35 db additional path loss.
(D) Terminal in NE Canada, Elevation Angle = 12° = > 1.05 db additional path loss.
(E) Small Terminal at GSFC, 76° W 40° N, Elevation Angle = 40° = > .47 db additional path loss.

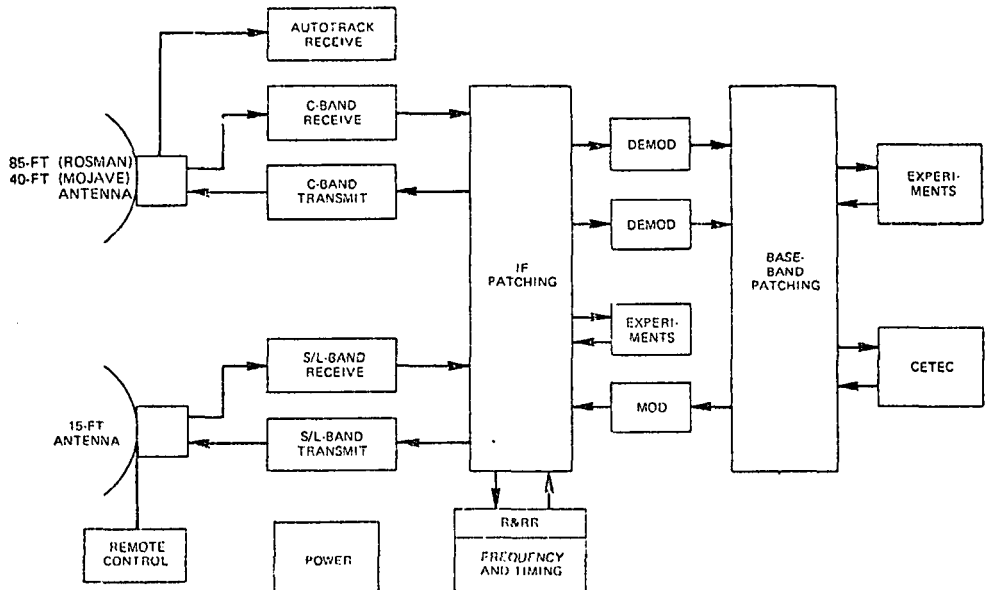


Figure VII 4. Rosman and Mojave Ground Station Block Diagram

RADIO FREQUENCY INTERFERENCE EXPERIMENT

The purpose of the RFIME is to provide realistic data on mutual RF interference between satellite and terrestrial telecommunications systems. This data is needed for optimum utilization of the frequency spectrum in the design and implementation of advanced communications satellite systems.

The C-Band RFIME is designed to measure and evaluate the effects of RF interference in the shared common carrier frequency band of 5925 to 6425 MHz.

The technical objectives of the C-Band RFIME are to:

- determine the flux density of 6 GHz interference power at the satellite
- establish practical gain-to-noise ratio limits for the satellite
- establish realistic satellite protection ratios
- determine both geographical and frequency distribution of terrestrial RF noise sources
- investigate the feasibility of establishing mathematical models for predicting RF interference.

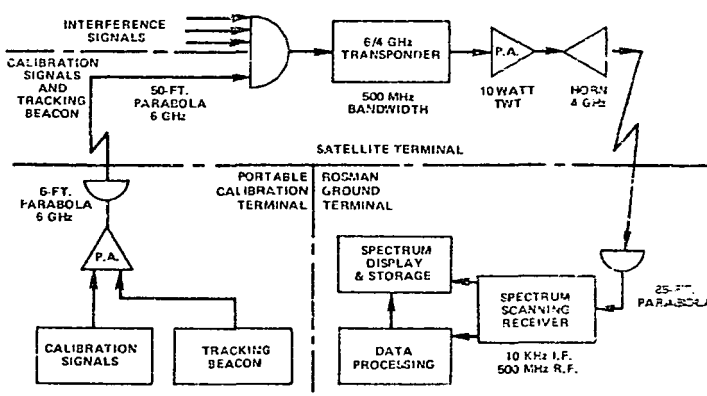


Figure VII-5. C-band RFI Experiment

The experiment measurement system configured as shown in Figure VII-8, utilizes a computer controlled, special purpose spectrum analyzer receiving system and data processor located at the Rosman ATS ground station. The small mobile terminal provides an effective means for calibrating the entire system.

MILLIMETER WAVE EXPERIMENT

The millimeter wave experiment is the second in a series of experiments designed to define the earth's space and near-earth propagation path characteristics at frequencies above 10 GHz in order to provide an extended spectrum for accomodating growth needs for the 1970's and beyond. The ATS-F experiment is designed to investigate atmospheric transmission characteristics on either side of the 22-GHz water-vapor absorption line, in regions which are proposed for use as earth-satellite communication channels. The object of the experiment is to obtain information on 20-GHz and 30-GHz satellite-to-earth transmissions for various earth stations and to establish a model for the communication channel in order to predict propagation effects under measurable climatic conditions.

The millimeter wave experiment operates in any of three modes: (1) a CW mode in which either or both of the millimeter wave signals will be continuously transmitted by the spacecraft in order to measure the attenuation for various atmospheric conditions, (2) a multitone mode in which either or both of the signals are generated along with known line-spectrum sidebands so that atmospheric effects on bandwidth and signal coherence may be studied and, (3) a communication data link mode in which a 6-GHz signal is transmitted to the satellite cross-strapped to the millimeter wave transmitter and retransmitted back to earth in order to compare the C-band and millimeter wave communication links.. The spacecraft equipment consists of multitone and CW signal generators, both 20-GHz and 30-GHz traveling wave tube (TWT) amplifiers and millimeter wave horn antennas.

It is desirable to have data from many ground stations operating under different weather conditions to provide maximum depth and validity to any conclusions which are reached; therefore, participation by others is encouraged. Detail specification are available through the Principal Investigator to all potential investigators. A plan for data acquisition, reduction, and analysis is also available. The Principal Investigator will act as the coordinator for all participants.

An extensive network of ground stations, augmented by ships and aircraft, support present day research and applications satellites. There are over 100 such fixed stations operated by or for agencies of the United States Government. Many of these stations have specialized capabilities and can only furnish useful support within the tracking and telemetry data transmission scheme for which they were designed. Therefore, the tracking and telemetry data on earth-orbiting satellites is acquired by a series of short observations and despite the large number of ground stations, many tracking and telemetry data relay gaps exist. (See Figure VII-5.) A single ground station at best can view only about 10 percent of the orbit of a satellite at 200 nautical miles altitude. This requires that orbit computations be based on sequential tracking by multiple ground stations and the storing of telemetry data on board the satellite then "dumping" the data at opportune times.

The tracking and data relay satellite system (T&DRSE) will employ earth-synchronous spacecraft to relay command, tracking and telemetry data between a few centrally located ground stations and multiple low earth-orbiting

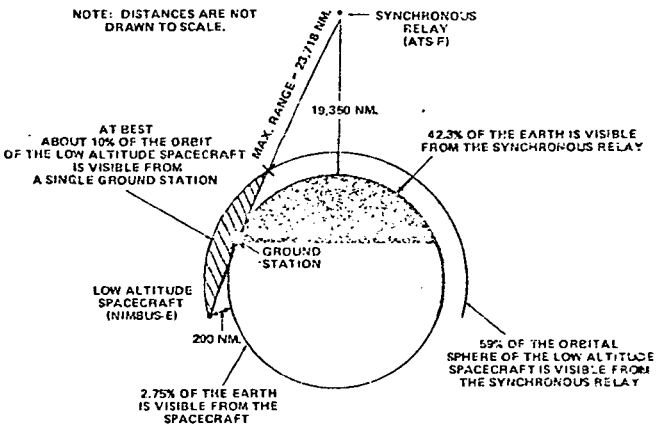

NOTE: DISTANCES ARE NOT DRAWN TO SCALE.
SYNCHRONOUS RELAY (ATS F)
MAX. RANGE - 23,718 NM.
19,350 NM.
AT BEST ABOUT 10% OF THE ORBIT OF THE LOW ALTITUDE SPACECRAFT IS VISIBLE FROM A SINGLE GROUND STATION
42.3% OF THE EARTH IS VISIBLE FROM THE SYNCHRONOUS RELAY
GROUND STATION
LOW ALTITUDE SPACECRAFT (NIMBUS-E)
200 NM.
59% OF THE ORBITAL SPHERE OF THE LOW ALTITUDE SPACECRAFT IS VISIBLE FROM THE SYNCHRONOUS RELAY
2.75% OF THE EARTH IS VISIBLE FROM THE SPACECRAFT

Figure VII-6. Visibility Contours

satellites. This concept makes it possible to view the low orbiting spacecraft for approximately 59% of the time from one ground station. Two such synchronous relays and ground stations properly placed could provide continuous coverage. Such a system would eliminate the need for tape recorders to store data on spacecraft and provide improved tracking data. This system can provide support for a wide range of space missions, including manned spacecraft. It can reduce the number of ground stations and provide a significant increase in capabilities.

The purpose of the experiment is to provide experience and information, using the ATS-F satellite, which can be used in designing future TDRS systems. The specific objectives are to:

- Determine the extent which the orbit of a low orbiting spacecraft can be established from another higher orbiting spacecraft
- Demonstrate the technology of command and telemetry data transmission between a low altitude satellite and a ground station using a geosynchronous satellite as a communications relay.

The tracking and data relay experiment uses the ATS-F as a repeater for retransmission of information between earth and a second satellite such as NIMBUS (See Figure VII-7). This experiment is a duplex link that requires the transponder to transmit and receive on two channels simultaneously. A signal originating on the ground is transmitted to ATS-F on C-band. The received ground signal is translated to IF, amplified, limited, and switched to two upconverters, C-band and S-band. The C-band signal is amplified and retransmitted to the ground for ranging and the S-band signal is amplified and transmitted to the satellite NIMBUS. Signals originating in the NIMBUS satellite are transmitted to ATS-F on S-band. In the transponder they are again translated to IF, amplified, and limited similar to the received ground signal. The processed signal is translated to a second C-band channel and transmitted to the ground. By phase locking the ATS frequency synthesizer to the C-band uplink signal, the output of the transponder is coherent with the ground transmitted signal. This allows direct Doppler rate ranging measurements of the ATS-F satellite signals and of the NIMBUS signals on the ground.

Single satellite tracking is accomplished by either closed loop monopulse tracking or open loop with computed pitch and roll commands based upon the ATS-F and low-altitude satellite ephemeris data. Before the actual low-altitude satellite transit occurs with open-loop tracking, the ground station must: (1) obtain suitably accurate orbit ephemeris data for the two satellites (low-altitude and ATS-F), (2) calculate the time-varying pitch and roll angles which the ATS-F

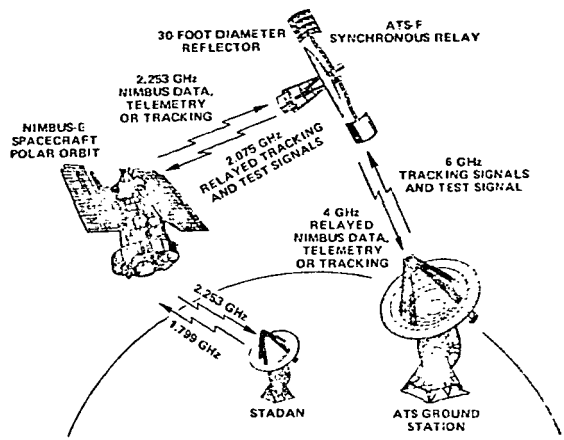

30 FOOT DIAMETER REFLECTOR
ATS-F SYNCHRONOUS RELAY
2.253 GHz NIMBUS DATA, TELEMETRY OR TRACKING
NIMBUS-E SPACECRAFT POLAR ORBIT
2.079 GHz RELAYED TRACKING AND TEST SIGNALS
6 GHz TRACKING SIGNALS AND TEST SIGNAL
4 GHz RELAYED NIMBUS DATA, TELEMETRY OR TRACKING
2.253 GHz
1.799 GHz
STADAN
ATS GROUND STATION

Figure VII-7. ATS-F/NIMBUS Data Relay System

must assume in order to point continuously at the low altitude satellite when it does transmit, (3) determine coefficients for the approximately polynominals for these time-periodic angle signals, (4) transmit these coefficients to the ATS-F Digital Operational Controller, (5) command the ATS-F to assume the initial values for the pitch and roll angles and, (6) the ground station must give a start-track signal at the appropriate time. When the S-band monopulse system is used, only items (5) and (6) of the above ground station requirements need to be considered.

The ATS-F spacecraft provide:

- Relay of two RF links
- Steerable S-band antenna beams capable of tracking a low-attitude satellite by maneuvering the ATS-F Z-axis
- The capability of tracking two low-altitude satellites with the S-band antenna beams, but only one beam must form part of an operational communication link

- Spacecraft fine pointing with an accuracy of 0.1 degree and a slewing rate of at least 17.5 degrees in thirty minutes
- Earth coverage C-band antenna beam
- Capability of using either coherent or noncoherent signal relaying
- Selectable IF noise bandwidths, 12 MHz and 40 MHz
- Relay of emergency voice from the NIMBUS satellite.

The ATS-F communications subsystem provides:

- Two selectable 3-db bandwidths, 12 MHz and 40 MHz, in the repeater
- Bandpass gain variation through the repeater at less than ± 0.5 db with respect to a 6-MHz signal frequency band centered in the 12-MHz bandpass, and with respect to a 20-MHz signal frequency band centered in the 40-MHz bandpass
- Repeater group delay over a chord covering the specified signal spectrum within the bandpasses that does not exceed:

Linear component:	0.30 nanoseconds/MHz
Parabolic component:	0.03 nanoseconds/MHz^2
Peak-to-peak ripple:	8.00 nanoseconds

- AM-to-PM conversion through the repeater that does not exceed 6 degrees per decibel
- C-band G/T of at least -20 db/° K and an EIRP of at least 24.5 dbw, both over a 25-degree conical field of view
- S-band G/T of at least 7.8/° K and an EIRP of at least 47.5 dbw over a 1-degree field of view; a G/T of at least 2.5 db/° K and an EIRP of at least 47.5 dbw over an angle of ± 7.5 degrees to both sides of the reflector axis and along the line traced by switching the beam centers.

SPACECRAFT ATTITUDE PRECISION POINTING & SLEWING ADAPTIVE CONTROL EXPERIMENT

The spacecraft attitude precision pointing and slewing adaptive control (SAPPSAC) experiment is designed to demonstrate a ground controlled real time feedback attitude control system for attitude maneuvering and precision pointing of the ATS-F spacecraft using onboard torquers and sensors with coupling through command and telemetry links.

The experiment will:

- Demonstrate the ability to maintain precise attitude stabilization of a given spacecraft pointing vector (such as the antenna) in a fixed direction for an extended period of time in the presence of all disturbing inputs when using the ground attitude control (GAC) command link with automatic execution and when using the normal command link for momentum wheel actuation with verification prior to execution

- Demonstrate relative command link reliabilities for the above period of time

- Demonstrate the relative attitude measurement capabilities for the available sensors during extended term precision attitude stabilization of the spacecraft

- Demonstrate the ability to perform a single attitude slewing maneuver between two reference ground locations in a manner prescribed [illegible] such as maneuver time, reaction jet propellant expenditure, momentum wheel speed changes, maximum allowable attitude rates, or combinations thereof)

- Demonstrate the ability to generate prescribed ground patterns such as antenna mapping at a ground station or fixed ground track generation

- Demonstrate the ability to track another object in flight in a manner that minimizes jet propellant expenditure or wheel speed variations

- Demonstrate the ability to meet precision pointing requirements in the presence of orbit changes (such as thruster firings) for a real time control environment

- Demonstrate the ability of the adaptive algorithms to accurately identify and track plant model parameters so that long term smoothing of sensor data can adequately maintain precise attitude stabilization

- Demonstrate the use of SAPPSAC for post-launch diagnostics such as verification of solar torque profiles, reaction jet behavior, momentum wheel behavior, low frequency jitter, and sensor behavior

- Demonstrate the ability to precisely align a given spacecraft pointing axis along the line-of-sight to any prescribed target reference

CESIUM BOMBARDMENT ION ENGINE EXPERIMENT

The primary objective of this experiment is to verify and obtain operational data on the utilization of an ion microthruster electric propulsion system. This propulsion system will be used for North-South station keeping orbital maneuvers. A secondary objective is the demonstration of thrust vectoring for attitude control.

The experiment flight hardware consists of two identical cesium bombardment ion microthruster systems. The two systems are mounted in the Earth Viewing Module (EVM) and aligned such that their undeflected thrust vectors pass through the spacecraft center of mass, normal to the roll axis. The assumed angle between the thrust vector and the yaw axis requires that the thrust capability of each microthruster be 1.0 mlb. Each microthruster system weighs about 33 lb and requires less than 150 watts of power.

ATS-F 13- AND 18-GHz PROPAGATION EXPERIMENT

Introduction

Occasional severe signal attenuation due to precipitation occurs when radio waves are transmitted through the atmosphere at frequencies above 10 GHz.

The theory of precipitation attenuation, although well understood, is only useful for simple atmospheric models as the large number of variables and the general lack of precise knowledge of climatological data prevents a theoretical definition of operational characteristics. Consequently, measurements must be made on a sufficiently large scale to encompass and define all of the situations likely to be encountered in a domestic or regional communications satellite system. The ATS-F 13- and 18-GHz Propagation Experiment is designed to make such measurements.

Objectives of the Experiment

The purpose of the experiment is to collect sufficient long-term data on propagation attenuation caused by precipitation for a large number of locations in the U.S. to permit determination of minimum power margins needed in spacecraft communications systems operating at frequencies above 10 GHz.

The experiment will involve the analysis of data gathered from 15 widely separated, dual-frequency sites, and will provide the unique capability of making instantaneous correlations of signals between any number of stations. To assist in evaluating the possibility of providing extra power for links suffering attenuation due to hydrometeors (rain drops) in the signal path, it will be necessary to determine how many stations simultaneously experience fades at a given fade level.

In addition, single-frequency transmitting stations will be used for the evaluation of the smaller scale (< 25 mile) spatial distribution of severe attenuation due to intense rainfall. The stations will be arranged to yield a number of separation distances. Correlation analyses of attenuation at these stations, taken two at a time, should reveal the separation with the lowest probability of simultaneous high attenuation over two satellite links.

It is expected that the duration of the experiment will be sufficiently long to permit a statistical comparison between the measured attenuation at a site and general meteorological parameters that are routinely collected by the weather bureau, such as rainfall rate, number of thunderstorm days, total precipitation, and weather radar information. This comparision is needed to provide a method for using the data collected in this experiment at other locations and to permit the extrapolation of the data to other periods at the selected location, since there is a great variation in weather conditions at any location from year to year.

To gather statistically significant data, the experiment will be conducted for the longest feasible period of time. The minimum acceptable duration of this experiment is estimated to be six months to include the months with greatest occurences of thunderstorms. During this period, the propagation experiments will be conducted on a 24-hour-a-day, seven-day-a-week basis. This is mandatory, since the widely scattered sites will be essentially unmanned. (No call up can be depended on for turning the experiment on when there is a prediction of precipitation for any site.) Furthermore, during rainy seasons, the probability of some rainfall occurring at any one site at any specific instant is reasonably high. Because the meteorological phenomenon of greatest importance to this propagation experiment is heavy precipitation during convective

thunderstorms and heavy frontal showers, the experiment must be conducted during the late spring to early fall period, when most of the precipitation (in the form of rain) occurs in the U. S.

Description of the System

The experimental system consists of three main parts:

1. Fifteen small, widely separated (> 100 miles apart) earth stations, each transmitting at approximately 13 and 18 GHz; and nine closely spaced (< 25 miles apart) 18-GHz transmitting terminals. These nine stations are arranged around three selected dual-frequency stations (Figure VII-8) in groups of three as shown in Figure VII-9.

2. A spacecraft transponder (Figure VII-10), receiving from the small earth transmitting terminals at approximately 13 and 18 GHz, and retransmitting these signals around 4 GHz.

3. One 4-GHz earth station for receiving and recording the propagation data from the ATS-F Propagation Experiment (Figure VII-11).

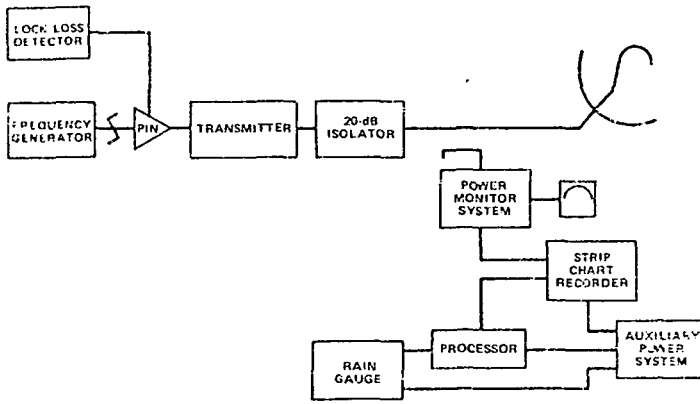


Figure VII-8. Transmitting Stations Block Diagram

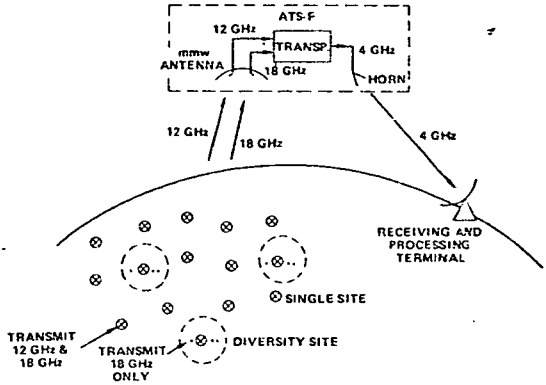


Figure VII-9. ATS-F Propagation Experiment System

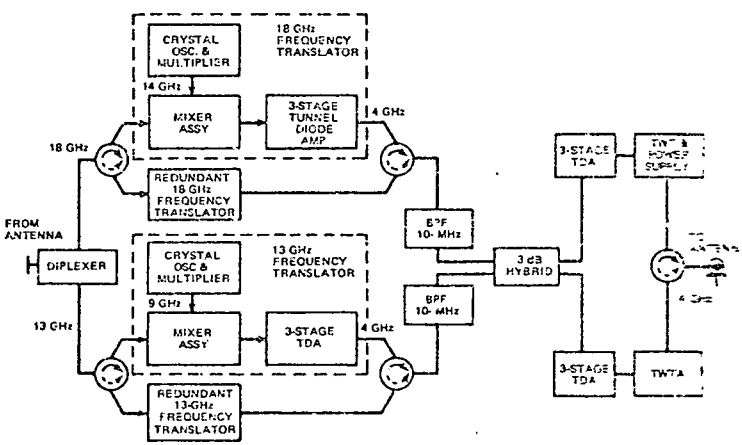


Figure VII-10. Transponder Block Diagram

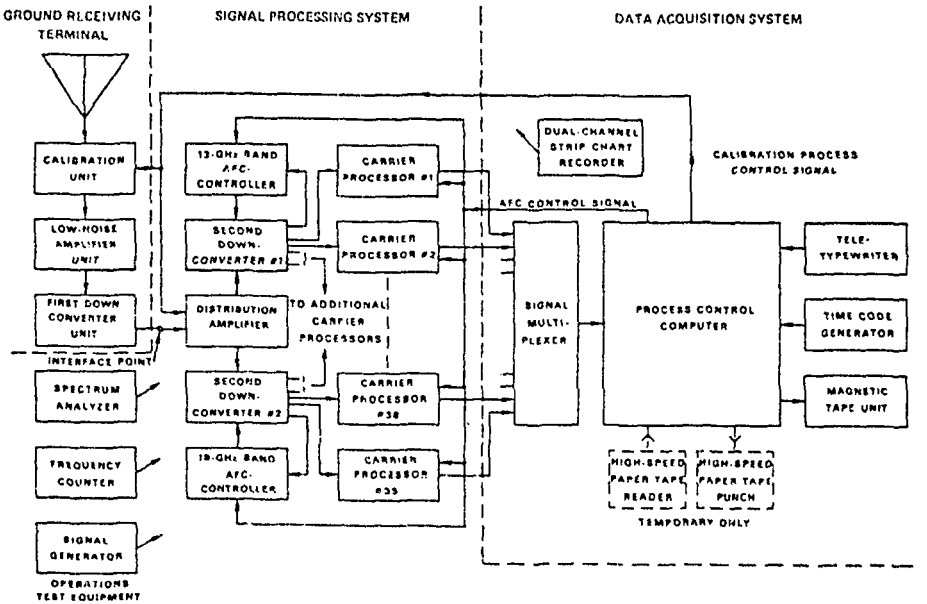


Figure VII-11. Receiving System Block Diagram

Actual operating frequencies are:

Uplink (small terminals to satellite)—Bands: 13.19-13.2 GHz and 17.79 -17.8 GHz

Downlink—Bands: 4140-4150 MHz and 4160-4170 MHz

Operation and Propagation Data

The experiment will essentially be performed from the main receiving station. The small earth transmitting stations will radiate a total of 39 CW carriers on the uplink to the spacecraft, 15 carriers at 13 GHz, and 24 carrier at 18 GHz. Each carrier will be on a slightly different frequency to allow separation, detection, and identification at the main earth terminal. These carriers will be received in the spacecraft, through a small millimeter wave antenna. Receivers operating at 13 and 18 GHz will frequency translate these carriers to approximately 4 GHz. They will then be linearly amplified in a TWTA and reradiated from an earth-coverage horn. The collection of carriers will be received at the main 4-GHz earth station, separated by a set of 39 phase-locked loops. The level of each carrier will then be synchronously AM detected, sampled, digitized, and stored on magnetic tapes for further processing.

The experimental package is designed for continuous operation. The package requires 14 watts of DC power and will occupy a total volume of approximately 0.75 cubic foot, and will weigh approximately 37 pounds. Operation during eclipse periods is not anticipated.

Link Power Budgets—Basic link power budgets are given in Table VII-8 for the uplink, and Table VII-9 for the downlink.

Table VII-8
Uplink Performance Parameters

	Attenuation Experiment		Diversity Experiment
	13 GHz	18 GHz	18 GHz
Transmitters			
Transmit Power, W (dbW)	25, (+14)	25, (+14)	5, (7)
Antenna Gain, dB	32	34.5	34.5
e.i.r.p., dBW	+46	48.5	41.5
Propagation Loss			
Path Loss, dB	-205.5	-209.1	-209.1
Clear Air Attenuation, dB	0.1	-0.7	-0.7
Total, dB	-205.6	-209.8	-209.8
Antenna Misalignment			
Transmitter, dB	-3.0	-4.0	-4.0
Satellite, dB	-1.5	-3.0	-3.0
Total, dB	-4.5	-7.0	-7.0
Satellite			
Antenna Gain, dB	25.8	28.6	28.6
Transmission Loss, dB	.6	.5	.6
Received Power (C), dBW	-138.9	-140.3	-147.8
System Noise Temperature, °K	2900	2900	2900
Noise Power Density (N_o), dB/Hz	-194	-194	-194
Up-Link Performance			
C/N_o	55.1	53.7	46.7

Table VII-9
4 GHz Downlink Performance Parameters

	Attenuation Experiment		Diversity Experiment
	13 GHz	18 GHz	18 GHz
Satellite			
Power per Carrier, dBW	-37.0	-28.9	-35.9
Antenna Gain, dB	17.0	17.0	17.0
Transmission Loss, dB	-1.0	-1.0	-1.0
e.i.r.p. per Carrier, dBW	-21.0	-12.9	-19.9
Propagation Loss			
(30° elevation) dB	-196	-196	-196
Antenna Misalignment			
Satellite, dB	-2.0	-2.0	-2.0
Earth Station			
Antenna Gain, dB	50.5	50.5	50.5
Received Power per Carrier (C), dBW	-168.5	-160.4	-167.4
System Noise Temperature, °K	80	80	80
System Noise Power Density (N_o), dBW/Hz	-209.6	-209.6	-209.6
Down-Link Performance			
C/N_o, dB/Hz	41.1	49.2	42.2
System Performance			
C/N_o Total, dB/Hz	41	47.9	40.9
Loop Bandwidth, Hz	100	100	100
SNR_L, dB	3	3	3
C/N_L, dB	18	24.9	17.9

Transponder Gain 101.9 @ 13 GHz
111.4 @ 18 GHz

LASER RETROREFLECTOR EXPERIMENT

Introduction

Since the first geostationary satellites were placed in orbit, the accuracy of satellite tracking systems has improved by at least two orders of magnitude. This improvement has been accomplished by such means as the recent advances in LASER technology, and two-way ranging at lunar distances; and has been demonstrated by the Polar Motion Experiment.

The Polar Motion Experiment, conducted at the Goddard Space Flight Center from June through October 1970 by Dr. D. E. Smith, has demonstrated that the internal consistencies of the LASERs are on the order of 30 to 40 centimeters. Preliminary orbit determination and intersite distance computations using these data are providing accuracies on the order of a few tens of centimeters. Recent advances in LASER ranging technology promise accuracies on the order of 5 to 10 centimeters within the next few years. Two-way ranging has already been accomplished at lunar distances using quartz cube-corner retroreflective arrays. The ATS-F and -G will carry LASER retroreflectors to provide for the conduct of experiments in LASER technology. Several geodetic experiments will be performed using the LASER data from the ATS-F spacecraft.

Objectives of the Experiment

The purpose of the LASER Retroreflector experiment is to (1) advance the technology for precisely determining the position of the ATS spacecraft along its orbit at any point in time, (2) redetermine the values of GM and geopotential coefficients, (3) investigate time dependence in the geopotential, and (4) conduct geodesy and earth physics experiments such as the measurement of polar motion, rotation of the earth, and solid earth tides.

ATS Position Determination Experiment—The purpose of this investigation is to improve the accuracy of position determinations for the ATS spacecraft through the use of the LASER data combined with data from the Goddard Range and Range Rate (GR&RR) tracking system. Several satellite-to-satellite experiments are planned using ATS-F and NIMBUS-E and ATS-F with GEOS-C.

The goals of the ATS-F/NIMBUS-E satellite-to-satellite tracking experiment are to (see Figure VII-12):

- Establish the extent to which an Earth orbiting satellite can be tracked by synchronous spacecraft

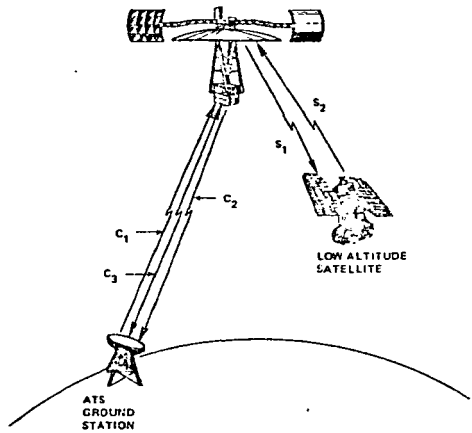


Figure VII-12. ATS-F Data Relay Experiment

- Compare the precision orbits of ATS-F and NIMBUS-E with orbits based upon conventional ground tracking techniques
- Link the results to the practical problems of future tracking and data relay satellite systems
- Determine to what extent gravitational model uncertainties and system errors can be separated
- Provide a basis for trade-off studies using combinations of synchronous satellite and conventional ground tracking

- Evaluate the effect of multipath and atmospheric distortions on the relay of tracking data and telemetry as a function of the NIMBUS-E geometry

- Establish the feasibility of eliminating onboard storage devices as well as evaluating command capability through ATS-F.

The objectives of the ATS-F/NIMBUS-E experiment and those of the ATS-F/GEOS-C experiment are quite similar with the addition of altimeter tracking on the GEOS-C. This will permit joint experiments and analyses to be conducted between the ATS-F and GEOS-C spacecraft involving the GEOS-C altimeter and satellite-to-satellite tracking. Correlations will be sought between altimeter measurements and satellite-to-satellite tracking data. Each of the two experiments has its own particular advantages and disadvantages.

A. Orbital Maneuvers

1. The NIMBUS-E orbit will be frequently perturbed by the firing of jets, thereby limiting the usefulness of some of the tracking data for geodetic purposes.

2. The GEOS-C orbit will be free of jet firing. This means that essentially all of the tracking data are potentially useful for geodetic purposes.

B. Other Tracking Coverage

1. The NIMBUS-E satellite will be tracked by ATS-F via its link and by the STADAN ground sites.

2. Tracking of GEOS-C, in addition to the ATS-F link, will be performed by optical, LASER, TRANET, (U.S. Navy's Doppler tracking system), C-band, unified S-band, and STADAN systems.

The orbit of GEOS-C will thus be known with geodetic accuracy on the basis of conventional ground tracking. This will permit evaluation of the ATS-F GEOS-C tracking data from the standpoints of geodesy and precise orbit determination.

C. Orbital Characteristics

1. NIMBUS-E will be in a near-polar orbit; thus, very nearly complete global-coverage sensing of the earth's gravitational field will be

obtained via the ATS-F/NIMBUS-E link. In addition, the ATS-F/NIMBUS-E experiment will provide tracking data while NIMBUS-E is traveling nearly perpendicular to the earth's equator.

2. GEOS-C will be in a rather low inclination orbit (about 20°). Tracking data will thus be obtained while the satellite is traveling approximately parallel to the equator. This will permit an additional type of sensing of the earth's gravitational field.

D. Tracking Geometry

1. NIMBUS-E can be tracked continuously all around its orbit approximately twice a day by ATS-F. Also, ATS-F will be able to provide tracking data both in, and perpendicular to, the orbital plane of NIMBUS-E. This coverage will strengthen the orbit determination solutions.

E. Technology

1. NIMBUS-E will receive the signal from ATS-F by means of an antenna installed on a steerable mount so that the antenna can point toward ATS-F.

2. GEOS-C will be equipped with a phased array which has the advantage of simplicity.

In each of the experiments, the accuracy with which the ATS satellite can be positioned in an inertial reference system directly affects their success. The key source of error is the uncertainty of the synchronous satellite location.

Specific error analysis studies were recently conducted for the ATS-F satellite. The studies indicated that the variations observed in the GR&RR position errors data could be removed by adding LASER tracking data and that the total position error could be resolved to the lowest observed GR&RR position error. This also occurred with height errors and, additionally, the greatest LASER tracking accuracy was in the radial direction.

Determination of GM and Geopotential Coefficients Experiment—The purpose of this experiment is to redetermine the values for GM and geopotential coefficients by combining the LASER tracking data from ATS-F and -G with the observations recorded from other satellites. The LASER data will provide continuous scale data; i.e., 10 to 15 minutes of data every hour, with an ac-

curacy in the tens of centimeters over time periods of several days. Since simultaneous tracking by several instruments will occur, it will be possible to define the position of the satellite through the geometry of the situation. This will tend to eliminate errors in the solution due to gravity model uncertainties. These uncertainties introduce approximately 50% of the total error in dynamic solutions. Solutions will also be obtained by imposing the orbital constraints for periods when simultaneous tracking is not available. These data will also be combined with Manned Space Flight and Jet Propulsion Laboratory radar data from deep space missions. The acquisition of tracking data during periods when the synchronous satellite will be experiencing a drift and also during the transfer trajectory phase of the mission will be of interest for determining low order resonant coefficients.

Geodesy and Earth Physics Experiment—Up to now geopotential coefficients have been derived through the analysis of satellite observations under the assumption that the geopotential is time independent. Due to the accuracy and distribution in time and space of the satellite observations it was not possible to investigate the possibility of time dependence in the geopotential. The geopotential does vary, however, whenever mass-displacements occur on or inside the earth such as tides, atmospheric motion, or earthquakes. The variations in the rotational rate of the earth are interpreted as evidence of these mass shifts. Kozai has identified a small annual variation in the J_2 coefficient of the geopotential. Kozai's results were based upon precision reduced Baker-Nunn optical observations with an accuracy on the order of 2 arc seconds (15 to 20 meters), whereas it is anticipated that the accuracy of the ATS LASER data will be in the tens of centimeters. The LASER data from the ATS-F and -G satellites will be combined with existing data from previously observed satellites. The stability of the ATS orbits will be quite valuable in this study since observations over time periods of several years are required in order to observe time dependence in the geopotential. Time dependence of longitude-dependent terms in the geopotential and their correlations with the westward drift of the geomagnetic field is thought to be a possible clue to mechanisms of core-mantle coupling. The accurate tracking data from these satellites will be investigated in an endeavor to detect not only periodic variations of J_2 and other coefficients including the gravitational constants GM, but also secular and irregular variations.

The motion of the earth's pole will be investigated through tracking data recorded from a LASER station at Rosman, North Carolina or Goddard Space Flight Center and a station in the southern hemisphere. The polar motion would be observed as a change in the computed inclination (Δ I) of the satellite as indicated in Figure VII-13.

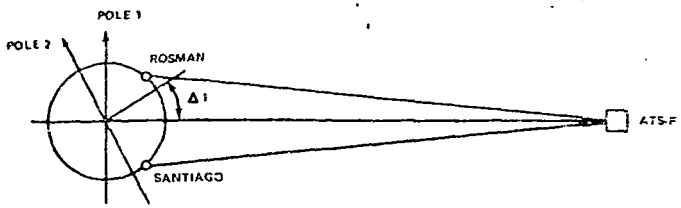


Figure VII-13. Polar Motion as a Change in Δ l

In conclusion, error analysis studies have indicated that the addition of LASER tracking to the ATS satellites will provide a significant reduction in the uncertainty of position determination for the satellites. This is especially true in the case of height uncertainties. The mobility of the LASER system is also an important factor for the ATS-F experiment where the satellites will be manoeuvered to different locations. Other experiments have also been outlined where the ATS data will be used in combination with data from other satellites for the recovery of geopotential constants and to study the time dependent geopotential. It should be pointed out that these goals and proposed studies will be reassessed and investigated in the light of any future technological advances prior to the launching of the ATS-F and -G satellites.

EDUCATIONAL TELEVISION EXPERIMENT

Introduction

The S-band educational television (ETV) experiment is a cooperative effort with the Corporation of Public Broadcasting (CPB), the Department of Health, Education and Welfare (HEW), and NASA. The experiment will evaluate the performance and effectiveness of satellite relay of educational programming to facilities such as schools, HEW learning centers, and community antenna television (CATV) distribution systems. ATS-F will be equipped with a two-channel TV transmit capability in the 2.5 to 2.69 MHz band recently allocated to the satellite broadcast service. Receiving stations at the various locations will be similar to the type of receiving station used for the existing instructional television fixed service (ITFS) now operating in this band.

System Capability

The capabilities and operating characteristics of the ATS-F ETV system are summarized in Table VII-10. High performance reception will be possible with ground receiving systems having at least 7-foot diameter fixed position parabolic antennas of the type currently in use in many ITFS installations throughout the country. Special receiving equipment including FM-to-AM converters will be required to handle the ETV transmissions from ATS-F. Consideration is being given to implementing limited "talk-back" capabilities to permit interactive and computer managed communications experiments using existing ATS-F communications capabilities at VHF and/or S-band (2250 MHz).

ATS-F Implementation

The basic ATS-F spacecraft includes a prime focus feed complex having a crossed-array of switchable broadband S-band feed elements (see Figure II-26). Two of these feed elements will be used for the ETV experiment. The two elements selected are the first and second offset elements in the south arm of the crossed array. With the spacecraft at 94° W longitude and utilizing several degrees of boresight pointing to the West of local nadir, these feeds will produce the approximate fields of view shown in Figure VII-14. This will provide ETV coverage to much of the Rocky Mountain area. Spacecraft boresight reorientation can be used to shift the fields of view to improve performance in regions to the East (Wyoming and Colorado) at the expense of coverage to the West (Idaho and Utah). No provision is being made for simultaneous operations in nonadjacent areas; therefore, operations outside the continental United States (Alaska or Hawaii) will have to be time shared with ETV operations within CONUS, with time allowed for moving the spacecraft boresight between transmissions.

ATS-F is being implemented to transmit ETV signals at two carrier frequencies within the 2500-2690 MHz band designated for satellite broadcast. The frequencies selected are 2569.2 MHz and 2670.0 MHz. Uplinks from ground to spacecraft will employ the existing C-band "work-horse" RF links at 5950, 6150, and/or 6350 MHz. Either ETV transmitter can drive either feed separately, and both transmitters can be connected to drive the same feed. Provision is made for monitoring either or both channels at C-band. No provision is made for a single transmitter driving both feeds simultaneously. Transmission of identical program material to adjacent areas is accomplished by routing a single ETV uplink signal to both transmitters connected to different feeds. Transmission of different program materials to the two areas of coverage requires that the spacecraft receive two uplink signals simultaneously.

Table VII-10
ETV Experiment Summary

Transmit Frequencies

Channel A - 2569.2 MHz
Channel B - 2670.0 MHz

Area Coverage

The adjacent spot areas ≥400 miles in diameter, positionable to serve CONUS, Alaska, and Hawaii on a time-shared basis. (See Figure VII-14.)

Transmit Power

≥15 watts, either channel

Signaling Format

Video modulation on an FM carrier (FM-TV); multiple sound channels on separate RF carriers.

Performance Objective

TASO Grade 1 (TV picture) equivalent when received at minimum 7-ft diameter antenna terminal.

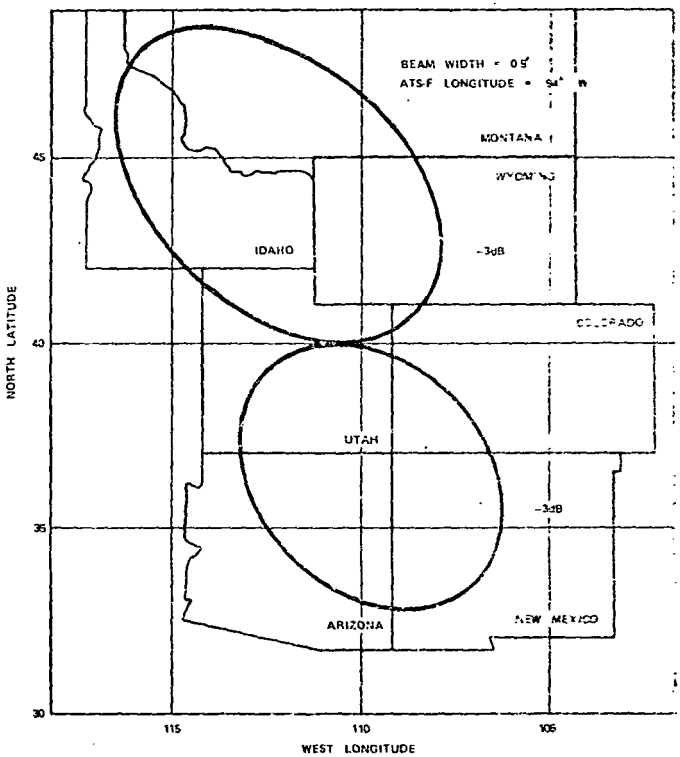


Figure VII-14. "Footprints" of ETV Coverage in Rocky Mountain States.

SECTION VIII

SCIENTIFIC EXPERIMENTS

Radio Beacon Experiment

Environmental Measurements Experiment

SECTION VIII

SCIENTIFIC EXPERIMENTS

RADIO BEACON EXPERIMENT

The radio beacon experiment provides a multifrequency spacecraft-to-ground radio link for investigating particles which affect radio propagation beyond the atmosphere.

There are three carrier frequencies: 40.0160 MHz, 140.0164 MHz and 360.1440 MHz. All radio signals are derived from a common oscillator. Amplitude modulation frequencies are 100 kHz (on the 40-MHz and 360-MHz signals) and 1000 kHz on the 40-MHz and 140-MHz signals. The phase reference for the modulation signals is the 100 kHz modulation on the 360.1440-MHz carrier.

The beacon is designed for several types of measurements, principally: Faraday rotation, differential phase (Doppler), phase and amplitude scintillation, and signal amplitude (absorption). Absorption measurements will have to be made on the 40-MHz signal. The other types of measurements are possible with the beacon as presently conceived. The mode of operation calls for continuous emission on all frequencies. There is no provision for intermittent operation unless a failure occurs nor for variation of frequencies. The overall design of the beacon is intentionally simple to improve the prospects of continuous operation and to eliminate the need to alert many observers to frequent changes in schedules.

All beacon signals are linearly polarized. Since the spacecraft is attitude controlled, it is possible to define the orientation of the emitted polarization. Differential Faraday measurements are possible using carriers and sidebands.

The phase stability of all modulation signals is ±1°. Correspondingly, when reorienting the satellite, the interferometric effect of displaced phase centers will not be greater than 1°. The absolute (transmitted) phase difference between the 360-MHz reference modulation and the modulation on any other carrier will not be more than 5° and will be known to within 1°. These phase stabilities are necessary for accurate determination of the integrated electron content by modulation phase delay (or content change by differential phase), and must be matched by equal precision of data recording and analysis. The carrier frequency stability is expected to be better than 2 parts in a 10^7 per year.

Effective radiated powers are sufficient for signal-to-noise (sky and receiver) ratios of 30 db with bandwidths on the order of 5 Hz and ground antennas with approximately 12 db gain. Wider bandwidths or lower gain reduces the S/N ratio and this will introduce phase fluctuations which impair the accuracy of the phase measurements.

The Principal Investigator, K. Davies of NOAA/ERL, will act as the coordinator of all participants to establish aspects of this experiment such as data interchange, standardization of data formats and special recording periods.

ENVIRONMENTAL MEASUREMENTS EXPERIMENT

The environmental measurements experiment (EME) package (Table VIII-1 is a group of eight experiments carried on board the ATS-F that are designed to study the spacecraft environment at synchronous altitude and to gain information on electromagnetic-ionospheric interactions. Six of the experiments are designed to obtain data on charged particles of several different types and over wide energy ranges. A seventh experiment is to provide magnetic field data, to be used in conjunction with the charged particle measurements to determine the dynamic processes which take place in the synchronous orbit environment.

The eighth experiment is a continuation of previous ATS studies into solar cell degradation.

Figure VIII-1 shows the EME and how the scientific experiments are integrated with it.

The EME will be operated during the entire mission. The scientific data obtained by the experiment will be recorded daily by the ATS ground stations with possible support by STADAN. Figure VIII-2 displays the operational configuration of the experiment. Data obtained will be recorded at the ground station, processed at GSFC, and sent to the individual experimenters. Also, attitude data and orbital ephemeris data will be provided to EME experimenters to aid in analysis of their data.

Low-Energy Proton-Electron

The low-energy proton-electron experiment measures electrons whose energy is from 2 kev to 25 kev in eleven differential energy windows and

Table VIII-1. Environmental Measurements Experiment

Experimenter	Experiment Characterization	Purpose*	Data				Prog. Sig.	Mode Cmd.	Wgt. (lb)	Avg. Power (watts)	Size N-) - Z (in.)
			Digital	Bi-Level	Analog	P. P.					
Univ. of New Hampshire	Low-Energy Proton-Electron	Make swept and pitched angle measurements of low energy electrons and protons in the magnetosphere.	8	0	0	4	5	9	7.0	5.0	9x6x6
NASA/GSFC	Low-Energy Proton	Study low energy electrons	3	0	2	7	9	9	4.7	1.6	7x9x4
McDonnell/ Douglas	Solar Cosmic Ray	Study solar cosmic rays, entry and propagation within magnetosphere, make detailed measurements of trapped electrons.	10	0	0	14	11	2	14.0	2.8	8x8x6
Univ. of Calif. at San Diego	Auroral Particles	Study the distribution of low energy electrons and protons.	5	14	0	15	11	30	9.5	6.0	6x9x6
Univ. of Minnesota	Electron-Proton Spectrometer	Investigate the origin of the Van Allen trapped radiation.	12	4	0	3	1	0	5.0	4.0	6x6x5
Univ. of Calif. at Los Angeles	Magnetometer	Study the magnetic field at synchronous distances.	0	1	6	3	2	0	8.3	4.5	8x8x6
Hughes	Solar Cell Radiation Damage	Measure the characteristics and performance degradation of solar cells in space	0	0	6	0	2	0	8.7	8.0	5-1/4x 9-1/2x 3
Aerospace	Omnidirectional Spectrometer	Measure omnidirectional flux and spectra of electrons and protons.	12	0	0	4	2	0	2.5	0.63	4x6x6
								Total	59.7	31.93	20x19x10

*Refer to text for detailed description.

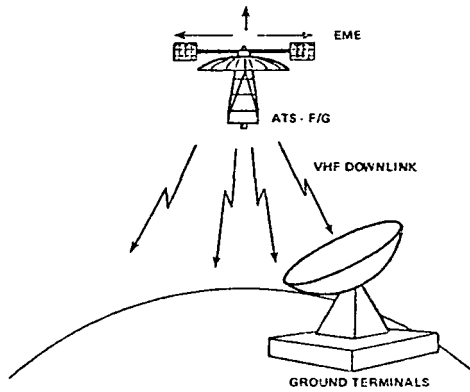


Figure VIII-1. ATS-F and -G/EME Operational Configuration

proton energy from 20 kev to 500 kev at two differential pitch angles. More specific objectives are:

(1) High-resolution electron measurements will be made for those particles that mirror near the equatorial plane.

(2) Make a two-point sample of the pitch angle distribution for electrons and protons once every second.

(3) Study local time dependence of observations.

(4) Study intensity time fluctuations of low-energy electrons and protons when an enhanced solar wind might push the subsolar magnetospheric boundary into the geostationary orbit.

(5) Correlation of the ATS data with ground, balloon, and rocket studies.

The experimenter is Roger L. Arnoldy of the Space Science Center at the University of New Hampshire.

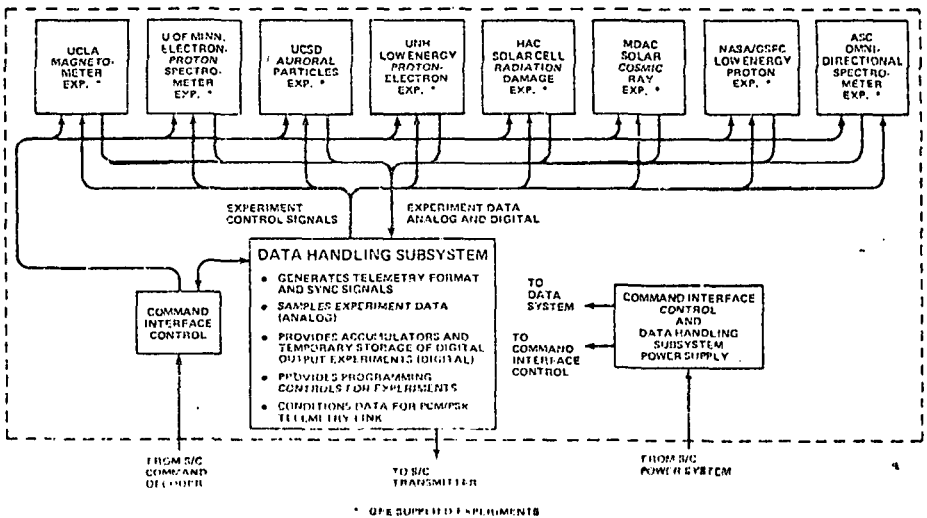

UCLA MAGNETO-METER EXP. *
U OF MINN. ELECTRON-PROTON SPECTRO-METER EXP. *
UCSD AURORAL PARTICLES EXP. *
UNH LOW ENERGY PROTON-ELECTRON EXP. *
HAC SOLAR CELL RADIATION DAMAGE EXP. *
MDAC SOLAR COSMIC RAY EXP. *
NASA/GSFC LOW ENERGY PROTON EXP. *
ASC OMNI-DIRECTIONAL SPECTRO-METER EXP. *
EXPERIMENT CONTROL SIGNALS
EXPERIMENT DATA ANALOG AND DIGITAL
DATA HANDLING SUBSYSTEM
• GENERATES TELEMETRY FORMAT AND SYNC SIGNALS
• SAMPLES EXPERIMENT DATA (ANALOG)
• PROVIDES ACCUMULATORS AND TEMPORARY STORAGE OF DIGITAL OUTPUT EXPERIMENTS (DIGITAL)
• PROVIDES PROGRAMMING CONTROLS FOR EXPERIMENTS
• CONDITIONS DATA FOR PCM/PSK TELEMETRY LINK
COMMAND INTERFACE CONTROL
TO DATA SYSTEM
TO COMMAND INTERFACE CONTROL
COMMAND INTERFACE CONTROL AND DATA HANDLING SUBSYSTEM POWER SUPPLY
FROM S/C COMMAND DECODER
TO S/C TRANSMITTER
FROM S/C POWER SYSTEM
* GFE SUPPLIED EXPERIMENTS

Figure VIII-2 AIEE Environmental Measurements Experiment

Low-Energy Protons

The low-energy proton experiment is an in-house experiment of the Goddard Space Flight Center. This experiment measures protons from 20 to 300 kev in 6 energy ranges. Also; it is the purpose of the experiment to answer several very important questions.

(1) Where in local time are the protons injected into the magnetosphere?

(2) Do the protons in the far outer belt follow L in their drift or are they affected by electric fields which permit them to enter and leave the magnetosphere?

(3) How closely in time is the injection of protons associated with auroral substorms?

Solar Cosmic Ray

The general objective of this experiment is to conduct a basic study of solar cosmic rays, their entry and propagation within the magnetosphere, and to measure detailed parameters of trapped electrons, both as functions of local time in orbit. Each measurement is performed at two different pitch angles. The measurements will be made with sufficient detail, energy resolution, dynamic intensity range, and time resolution to allow several studies during the period following maximum solar cosmic ray activity. Some of the specific objectives are to:

(1) Investigate the mechanism by which low-energy solar cosmic rays gain entry and propagate within the magnetosphere.

(2) Investigate the mechanism responsible for cutoff variations during geomagnetic activity.

(3) Use the data as a test to evaluate and to improve a recently developed magnetospheric model which includes the tilt of the dipole axis to the solar wind.

(4) Compare the time history and development of solar cosmic ray events at L = 6.6Re to the Douglas observatories in the Northern and Southern Polar regions.

(5) Study the behavior of trapped electrons as a function of local time and geomagnetic activity in the ATS orbit.

(6) Directly compare spectral and intensity changes between two families of particles which traverse vastly different regions of the magnetosphere in order to investigate acceleration and propagation processes during geomagnetic activity and during quiet periods.

Energy ranges are, for alpha particles 1.2 Mev to 180 Mev, for protons 0.2 Mev to 300 Mev and for electrons 50 kev to 800 kev.

The experimenter is A. J. Masley of the Space Sciences Department at the McDonnell-Douglas Aircraft Corporation, Huntington Beach, California.

Auroral Particles

The scientific objectives of the experiment are as follows:

(1) Map the distribution of low energy (50 ev - 50 kev) electrons and protons on a constant line of force so that correlation studies between these particle fluxes and the visible aurora can be conducted to determine the nature of the accelerating mechanism in the magnetosphere.

(2) Make studies of the rapid time variation of the shape of the spectra and location of any peaks in the particle distribution.

(3) Compare ATS-F data with that from other spacecraft.

(4) Make detailed studies of the energy balance between particles and fields at their location and relate it to the magnetosphere as a whole.

The experimenter is Carl E. Mc Ilwain of the Space Science Laboratories at the University of California at San Diego.

Particle Acceleration Measurements

The objective of the University of Minnesota experiment is to investigate the origin of the Van Allen trapped radiation. Measurements of the intensity

and time variations of protons and electrons in the vicinity of the synchronous orbit, and detailed analysis of these variations in their relationships to polar and magnetic storms and other perturbations of the magnetosphere will further the knowledge of magnetospheric dynamics of acceleration and modulation processes in several ways:

(1) Establish, in a more direct manner, the existence or nonexistence at the geostationary orbit of certain rapidly varying electron fluxes commonly observed in precipitated fluxes.

(2) Study the role the protons of these energies play in the dynamics that produce large fluxes of energetic electrons.

(3) Study the pitch angle distribution characteristics of protons and electrons during the various modulated activities, such as the quasi-periodic pulsation activity.

(4) Study more precisely the dynamic variables that may be affected in transit from the equator to the auroral zone.

Energy ranges are for protons 20 kev to 500 kev and for electrons 20 to 40 kev, 100 to 200 kev, and 1 to 1.5 Mev.

The experimenter is John R. Winkler of the School of Physics and Astronomy at the University of Minnesota.

Magnetometer

The primary objective of the UCLA experiment is the study of the magnetic field at synchronous distances along the three orthogonal axes of the spacecraft. The specific goals are as follows:

(1) Study the properties of various magnetohydrodynamic phenomena in the magnetosphere, the magnetospheric tail, and the magnetosheath.

(2) Study the interaction of the solar wind with the magnetosphere and the magnetospheric tail.

(3) Study the properties of the magnetospheric boundary or magnetopause.

(4) Study the properties of the magnetosheath.

The experimenter is Paul J. Coleman, Jr., of the Space Science Center of the University of California at Los Angeles.

Solar Cell Radiation Damage

The objective of this experiment is directed primarily toward isolating the predominant degradation mechanism(s) associated with current production solar cells and eliminating anomalous data through increased data points and instrumentation accuracy. A total of 80 solar cells will be individually monitored on the flight experiment with a total of 12 current-voltage points and temperature data for each solar cell transmitted to ground on a real-time basis. Five solar cells of 16 types have been included to provide a statistically meaningful sample size.

The experimenter is W. Dunkerly of the Space Systems Division at the Hughes Aircraft Company, El Segundo, California.

Omnidirectional Spectrometer

The purpose of this experiment is to measure the omnidirectional fluxes and spectra of electron and protons. Four detectors will count protons from 2 or 3 to 10 Mev, 10 to 21 Mev, 20 to 40 Mev, and 40 to 80 Mev. They will also count, respectively, electrons of energies greater than 80 kev and 250 kev, 600 kev, 1.2 Mev, and greater than 4 Mev.

The experimenter is G. A. Paulikas, Aerospace Corporation, Los Angeles, California.

APPENDIX A

USEFUL INFORMATION

APPENDIX A

USEFUL INFORMATION

Presented below and on the following pages are tables, charts, and instructions that may prove useful to the reader in interpreting information contained in the ATS-F Data Book.

The following is a listing of the data presented:

Frequently used conversion factors and constants
Names of International Units
Definitions of International Units
Prefixes
Physical constants
Conversion factors
A table containing conversion factors for terms now becoming obsolete
Radio frequency spectrum pertinent to ATS-F and -G
Instructions for determining antenna azimuth and elevation angles to any equatorial synchronous satellite
Plot for determination of ground station azimuth and elevation angles to any equatorial synchronous satellite
Time conversion chart
Elevation angles for visibility with a satellite at synchronous altitude, 0 latitude and longitude shown; from the listed stations
24-hour satellite longitudinal drift acceleration vs longitude
ATS-F transmission links data

TABLES AND CHARTS

Frequently Used Conversion Factors and Constants

1 Statute Mile = 1.60935 Kilometers
1 Statute Mile = 0.8689779 International Nautical Miles
1 Kilometer = 0.62137 Statute Miles
1 Kilometer = 0.5399568 International Nautical Miles
1 International Nautical Mile = 6076.1033 Feet = 1.150777 Statute Miles
1 International Nautical Mile = 1852 Meters = 1.852 Kilometers

1 Radian = 57.29578 Degrees
1 Degree = 0.0174532923 Radian
J = 1.6232×10^{-3}
2π = 6.2831853

Useful Constants

Velocity of light in a vacuum

2.99796×10^{10} cm/sec
983.571×10^{6} ft/sec
186,284 statute miles/sec
162×10^{3} nautical miles/sec

Nominal synchronous altitude (above the earth surface)

22,236 statute miles
19,323 nautical miles
35,781 kilometers
11.740×10^{7} feet

Synchronous, equatorial velocity

10,060 feet/second
3,066.6 meters/second

Physical Quantity	*Name of Unit*	*Symbol*	
	BASIC UNITS		
Length	meter	m	
Mass	kilogram	kg	
Time	second	s	
Electric current	ampere	A	
Temperature	kelvin	K	
Luminous intensity	candela	cd	
	DERIVED UNITS		
Area	square meter	m^2	
Volume	cubic meter	m^3	
Frequency	hertz	Hz	(s^{-1})
Density	kilogram per cubic meter	kg/m^3	
Velocity	meter per second	m/s	
Angular velocity	radian per second	rad/s	
Acceleration	meter per second squared	m/s^2	
Angular acceleration	radian per second squared	rad/s^2	
Force	newton	N	($kg \cdot m/s^2$)
Pressure	newton per sq meter	N/m^2	
Kinematic viscosity	sq meter per second	m^2/s	
Dynamic viscosity	newton-second per sq meter	$N \cdot s/m^2$	
Work, energy, quantity of heat	joule	J	($N \cdot m$)
Power	watt	W	(J/s)
Electric charge	coulomb	C	($A \cdot s$)
Voltage, potential difference, electromotive force	volt	V	(W/A)
Electric field strength	volt per meter	V/m	
Electric resistance	ohm	Ω	(V/A)
Electric capacitance	farad	F	($A \cdot s/V$)
Magnetic flux	weber	Wb	($V \cdot s$)
Inductance	henry	H	($V \cdot s/A$)
Magnetic flux density	tesla	T	(Wb/m^2)
Magnetic field strength	ampere per meter	A/m	
Magnetomotive force	ampere	A	
Luminous flux	lumen	lm	(cd sr)
Luminance	candela per sq meter	cd/m^2	
Illumination	lux	lx	(lm/m^2)
Wave number	1 per meter	m^{-1}	
Entropy	joule per kelvin	J/K	
Specific heat	joule per kilogram kelvin	$J \cdot kg^{-1} \cdot K^{-1}$	
Thermal conductivity	watt per meter kelvin	$W \cdot m^{-1} \cdot K^{-1}$	
Radiant intensity	watt per steradian	W/sr	
Activity (of a radioactive source)	1 per second	s^{-1}	
	Supplementary Units		
Plane angle	radian	rad	
Solid angle	steradian	sr	

Definitions of International Units

Definitions of the most important SI Units are given in the following paragraphs. These definitions have been extracted from the records of the International Committee and the General Conferences.

meter (m)

The *meter* is the length equal to 1 650 763.73 wavelengths in vacuum of the radiation corresponding to the transition between the levels $2\ p_{10}$ and $5\ d_5$ of the krypton-86 atom.

kilogram (kg)

The *kilogram* is the unit of mass; it is equal to the mass of the international prototype of the kilogram. (The international prototype of the kilogram is a particular cylinder of *platinum-iridium alloy* which is preserved in a vault at Sèvres, France, by the International Bureau of Weights and Measures.)

second (s)

The *second* is the duration of 9 192 631 770 periods of the radiation corresponding to the transition between the two hyperfine levels of the ground state of the cesium-133 atom.

ampere (A)

The *ampere* is that constant current which, if maintained in two straight parallel conductors of infinite length, of negligible circular cross section, and placed 1 meter apart in vacuum, would produce between these conductors a force equal to 2×10^{-7} newton per meter of length.

kelvin (K)

The *kelvin*, unit of thermodynamic temperature, is the fraction 1/273.16 of the thermodynamic temperature of the triple point of water.

candela (cd)

The *candela* is the luminous intensity, in the perpendicular direction, of a surface of 1,600 000 square meter of a blackbody at the temperature of freezing platinum under a pressure of 101 325 newtons per square meter.

newton (N)

The *newton* is that force which gives to a mass of 1 kilogram an acceleration of 1 meter per second per second.

joule (J)

The *joule* is the work done when the point of application of 1 newton is displaced a distance of 1 meter in the direction of the force.

watt (W)

The *watt* is the power which gives rise to the production of energy at the rate of 1 joule per second.

volt (V)

The *volt* is the difference of electric potential between two points of a conductor when carrying a constant current of 1 ampere, when the power dissipated between these points is equal to 1 watt.

ohm (Ω)

The *ohm* is the electric resistance between two points of a conductor when a constant difference of potential of 1 volt, applied between these two points, produces in this conductor a current of 1 ampere, this conductor not being the source of any electromotive force.

coulomb (C)

The *coulomb* is the quantity of electricity transported in 1 second by a current of 1 ampere.

farad (F)

The *farad* is the capacitance of a capacitor between the plates of which appears a difference of potential of 1 volt when it is charged by a quantity of electricity equal to 1 coulomb.

Definitions of International Units (Cont)

weber (Wb)

The *weber* is the magnetic flux which, linking a circuit of one turn, produces in it an electromotive force of 1 volt as it is reduced to zero at a uniform rate in 1 second.

henry (H)

The *henry* is the inductance of a closed circuit in which an electromotive force of 1 volt is produced when the electric current in the circuit varies uniformly at a rate of 1 ampere per second.

lumen (lm)

The *lumen* is the luminous flux emitted in a solid angle of 1 steradian by a uniform point source having an intensity of 1 candela.

Prefixes

The names of multiples and submultiples of SI Units may be formed by application of the prefixes:

Factor by which unit is multiplied	Prefix	Symbol
10^{12}	tera	T
10^{9}	giga	G
10^{6}	mega	M
10^{3}	kilo	k
10^{2}	hecto	h
10	deka	da
10^{-1}	deci	d
10^{-2}	centi	c
10^{-3}	milli	m
10^{-6}	micro	μ
10^{-9}	nano	n
10^{-12}	pico	p
10^{-15}	femto	f
10^{-18}	atto	a

Physical Constants

The following lists of physical constants are from the work of B. N. Taylor, W. H. Parker and D. N. Langenberg (*Reviews of Modern Physics*, July 1969). *Their least-squares adjustment of values of the constants depends strongly on a new and highly accurate (2.4 ppm) determination of e/h from the ac Josephson effect in superconductors, and is believed to be more accurate than the 1965 adjustment which appears to suffer from the use of an incorrect value of the fine structure constant as an input datum.*

Quantity	Symbol	Value	Error ppm	Prefix	Unit
Speed of light in vacuum	c	2 997 925 0	0. 33	$\times 10^{8}$	m s^{-1}
Gravitational constant	G	6. 673 2	460	10^{-11}	N m^2 kg^{-2}
Avogadro constant	N_A	6. 022 169	6. 6	10^{26}	kmole^{-1}
Boltzmann constant	k	1. 380 622	43	10^{-23}	J K^{-1}
Gas constant	R	8. 314 34	42	10^{3}	J kmole^{-1} K^{-1}
Volume of ideal gas, standard conditions	V_0	2. 241 36	------	10^{1}	m^3 kmole^{-1}
Faraday constant	F	9. 648 670	5. 5	10^{7}	C kmole^{-1}
Unified atomic mass unit	u	1. 660 531	6. 6	10^{-27}	kg
Planck constant	h	6. 626 196	7. 6	10^{-34}	J s
	$h/2\pi$	1. 054 591 9	7. 6	10^{-34}	J s
Electron charge	e	1. 602 191 7	4. 4	10^{-19}	C
Electron rest mass	m_e	9. 109 558	6. 0	10^{-31}	kg
		5. 485 930	6. 2	10^{-4}	u
Proton rest mass	m_p	1. 672 614	6. 6	10^{-27}	kg
		1. 007 276 61	. 03	------	u
Neutron rest mass	m_n	1. 674 920	6. 6	10^{-27}	kg
		1. 008 665 20	. 10	------	u
Electron charge to mass ratio	e/m_e	1. 758 802 6	3. 1	10^{11}	C kg^{-1}
Stefan-Boltzmann constant	σ	5. 669 61	170	10^{-8}	W m^{-2} K^{4}
First radiation constant	$8\pi hc$	4. 992 579	7. 6	10^{-16}	J m
Second radiation constant	hc/k	1. 438 533	43	10^{-2}	m K
Rydberg constant	R_∞	1. 097 373 12	. 10	10^{7}	m^{-1}
Fine structure constant	α	7. 297 351	1. 5	10^{-3}	
	α^{-1}	1. 370 360 2	1. 5	10^{2}	
Bohr radius	a_0	5. 291 771 5	1. 5	10^{-11}	m
Classical electron radius	r_e	2. 817 939	4. 6	10^{-15}	m
Compton wavelength of electron	λ_C	2. 426 309 6	3. 1	10^{-13}	m
	$\lambda_C/2\pi$	3. 861 592	3. 1	10^{-13}	m
Compton wavelength of proton	$\lambda_{C,p}$	1. 321 440 9	6. 8	10^{-15}	m
	$\lambda_{C,p}/2\pi$	2. 103 139	6. 8	10^{-16}	m
Compton wavelength of neutron	$\lambda_{C,n}$	1. 319 621 7	6. 8	10^{-15}	m
	$\lambda_{C,n}/2\pi$	2. 100 243	6. 8	10^{-16}	m
Electron magnetic moment	μ_e	9. 284 851	7. 0	10^{-24}	J T^{-1}
Proton magnetic moment	μ_p	1. 410 620 3	7. 0	10^{-26}	J T^{-1}
Bohr magneton	μ_B	9. 274 096	7. 0	10^{-24}	J T^{-1}
Nuclear magneton	μ_n	5. 050 951	10	10^{-27}	J T^{-1}

Physical Constants (Cont)

Quantity	Symbol	Value	Error ppm	Prefix	Unit
Gyromagnetic ratio of protons in H_2O	γ'_p	2.675 127 0	3.1	10^8	rad s^{-1} T^{-1}
	$\gamma'_p/2\pi$	4.257 597	3.1	10^6	Hz T^{-1}
Gyromagnetic ratio of protons in H_2O corrected for diamagnetism of H_2O	γ_p	2.675 156 5	3.1	10^8	rad s^{-1} T^{-1}
	$\gamma_p/2\pi$	4.257 707	3.1	10^6	Hz T^{-1}
Magnetic flux quantum	Φ_0	2.067 553 8	3.3	10^{-15}	Wb
Quantum of circulation	$h/2m_e$	3.636 947	3.1	10^{-4}	J s kg^{-1}
	h/m_e	7.273 894	3.1	10^{-4}	J s kg^{-1}

"Dimensionless" combination	Value	Error ppm	Prefix
[illegible]eV	5.099 538	4.4	10^{33}
[illegible]	9 314 512	5.5	10^{9}
[illegible]	1.660 531	6.6	10^{-27}
[illegible]	5.110 041	3.1	10^{5}
[illegible]	9.382 592	5.5	10^{8}
[illegible]	9.365 527	5.5	10^{7}
[illegible] J	1.602 101 7	4.4	10^{-19}
[illegible] H_2	2.417 965 9	3.3	10^{14}
[illegible] m	8.065 465	3.3	10^{3}
[illegible] K	1.160 455	42	10^{4}
[illegible]	1.239 554 1	3.3	10^{-6}
[illegible] J	2.179 914	7.6	10^{-18}
[illegible]	1.360 582 6	3.3	10^{-7}
[illegible] Hz	3.289 842 3	.35	10^{15}
[illegible] K	1.573 936	43	10^{5}
[illegible]	1.836 109	6.2	10^{3}
[illegible]	1.001 152 628 9	.0031	
[illegible]	1.520 995 72	.066	10^{-3}
[illegible]	1.521 032 04	.30	10^{-3}
[illegible]	2.792 709	6.2	
[illegible]	2.792 782	6.2	

The following tables express the definitions of miscellaneous units of measure as exact numerical multiples of coherent SI Units, and provide multiplying factors for converting numbers and miscellaneous units to corresponding new numbers and SI Units.

The first two digits of each numerical entry represent a power of 10. An asterisk follows each number which expresses an exact definition. For example, the entry "−02 2.54*" expresses the fact that 1 inch=2.54×10^{-2} meter, exactly, by definition. Most of the definitions are extracted from National Bureau of Standards documents. Numbers not followed by an asterisk are only approximate representations of definitions, or are the results of physical measurements.

The conversion factors are listed alphabetically and by physical quantity.

The Listing by Physical Quantity includes only relationships which are frequently encountered and deliberately *omits the great multiplicity of combinations of units which are used for more* specialized purposes. Conversion factors for combinations of units are easily generated from numbers given in the Alphabetical Listing by the technique of direct substitution or by other well-known rules for manipulating units. These rules are adequately discussed in many science and engineering textbooks and are not repeated here.

Alphabetical Listing

To convert from	*to*	*multiply by*
abampere	ampere	+01 1.00*
abcoulomb	coulomb	+01 1.00*
abfarad	farad	+09 1.00*
abhenry	henry	−09 1.00*
abmho	mho	+09 1.00*
abohm	ohm	−09 1.00*
abvolt	volt	−08 1.00*
acre	meter^2	+03 4.046 856 422 4*
ampere (international of 1948)	ampere	− 01 9.998 35
angstrom	meter	−10 1.00*
are	meter^2	+02 1.00*
astronomical unit	meter	+11 1.495 978 9
atmosphere	newton/meter^2	+05 1.013 25*
bar	newton/meter^2	+05 1.00*
barn	meter^2	−28 1.00*
barrel (petroleum, 42 gallons)	meter^3	− 01 1.589 873
barye	newton/meter^2	− 01 1.00*
British thermal unit (ISO/TC 12)	joule	+03 1.055 06
British thermal unit (International Steam Table)	joule	+03 1.055 04
British thermal unit (mean)	joule	+03 1.055 87
British thermal unit (thermochemical)	joule	+03 1.054 350 264 488
British thermal unit (39° F)	joule	+03 1.059 67
British thermal unit (60° F)	joule	+03 1.054 65
bushel (U.S.)	meter^3	−02 3.523 907 016 688*

To convert from	to	multiply by
cable	meter	+02 2.194 56*
caliber	meter	−04 2.54*
calorie (International Steam Table)	joule	+00 4.1868
calorie (mean)	joule	+00 4.190 02
calorie (thermochemical)	joule	+00 4.184*
calorie (15° C)	joule	+00 4.185 50
calorie (20° C)	joule	+00 4.181 90
calorie (kilogram, International Steam Table)	joule	+03 4.1868
calorie (kilogram, mean)	joule	+03 4 190 02
calorie (kilogram, thermochemical)	joule	+03 4.184*
carat (metric)	kilogram	−04 2.00*
Celsius (temperature)	kelvin	$t_K = t_C + 273\ 15$
centimeter of mercury (0° C)	newton/meter²	+03 1.333 22
centimeter of water (4° C)	newton/meter²	+01 9.806 38
chain (engineer or ramden)	meter	+01 3.045*
chain (surveyor or gunter)	meter	+01 2.011 68*
circular mil	meter²	−10 5.067 074 8
cord	meter³	+00 3.624 555 3
coulomb (international of 1948)	coulomb	−01 9.998 35
cubit	meter	−01 4.572*
cup	meter³	−04 2.365 882 365*
curie	disintegration/second	+10 3.70*
day (mean solar)	second (mean solar)	+04 8.64*
day (sidereal)	second (mean solar)	+04 8.616 409 0
degree (angle)	radian	−02 1.745 329 251 994 3
denier (international)	kilogram/meter	−07 1.00*
dram (avoirdupois)	kilogram	−03 1.771 845 195 312 5*
dram (troy or apothecary)	kilogram	−03 3.887 934 6*
dram (U.S. fluid)	meter³	−06 3.696 691 195 312 5*
dyne	newton	−05 1.00*
electron volt	joule	−19 1.602 10
erg	joule	−07 1.00*
Fahrenheit (temperature)	kelvin	$t_K = (5/9)\ (t_F + 459.67)$
Fahrenheit (temperature)	Celsius	$t_C = (5/9)\ (t_F - 32)$
farad (international of 1948)	farad	−01 9.995 05
faraday (based on carbon 12)	coulomb	+04 9.648 70
faraday (chemical)	coulomb	+04 9.649 57
faraday (physical)	coulomb	+04 9.652 19
fathom	meter	+00 1.828 8*
fermi (femtometer)	meter	−15 1.00*
fluid ounce (U.S.)	meter³	−05 2.957 352 956 25*
foot	meter	−01 3.048*
foot (U.S. survey)	meter	+02 1200/3937
foot (U.S. survey)	meter	−01 3.048 006 096
foot of water (39.2° F)	newton/meter²	+03 2.988 98
foot-candle	lumen/meter²	+01 1.076 391 0
foot-lambert	candela/meter²	+00 3.426 259
furlong	meter	+02 2.011 68*
gal (galileo)	meter/second²	−02 1.00*
gallon (U.K. liquid)	meter³	−03 4.546 087
gallon (U.S. dry)	meter³	−03 4.404 883 770 86*
gallon (U.S. liquid)	meter³	−03 3.785 411 784*

Conversion Factors (Cont.)

To convert from	to	multiply by
gamma	tesla	10^{-9} 1.00*
gauss	tesla	10^{-4} 1.00*
gilbert	ampere turn	10^{-1} 7.957 747 2
gill (U.K.)	meter³	10^{-4} 1.420 653
gill (U.S.)	meter³	10^{-4} 1.182 941 2
grad	degree (angular)	10^{-1} 9.00*
grad	radian	10^{-2} 1.570 796 3
grain	kilogram	10^{-5} 6.479 891*
gram	kilogram	10^{-3} 1.00*
hand	meter	10^{-1} 1.016*
hectare	meter²	10^{4} 1.00*
henry (international of 1948)	henry	10^{0} 1.000 495
hogshead (U.S.)	meter³	10^{-1} 2.384 809 420 338 92*
horsepower (550 foot lbf/second)	watt	10^{2} 7.456 998 7
horsepower (boiler)	watt	10^{3} 9 809 50
horsepower (electric)	watt	10^{2} 7.46*
horsepower (metric)	watt	10^{2} 7.354 99
horsepower (U.K.)	watt	10^{2} 7.457
horsepower (water)	watt	10^{2} 7.460 43
hour (mean solar)	second (mean solar)	10^{3} 3.60*
hour (sidereal)	second (mean solar)	10^{3} 3.590 170 4
hundredweight (long)	kilogram	10^{1} 5.080 234 544*
hundredweight (short)	kilogram	10^{1} 4.535 9[illegible] 7*
inch	meter	10^{-2} 2.54*
inch of mercury (32° F)	newton/meter²	10^{3} 3.386 [illegible]
inch of mercury (60° F)	newton/meter²	10^{3} 3.37[illegible] 53
inch of water (39.2° F)	newton/meter²	10^{2} 2.490 82
inch of water (60° F)	newton/meter²	10^{2} 2.4884
joule (international of 1948)	joule	10^{0} 1.000 165
kayser	1/meter	10^{2} 1.00*
kilocalorie (International Steam Table)	joule	10^{3} 4.186 74
kilocalorie (mean)	joule	10^{3} 4.190 02
kilocalorie (thermochemical)	joule	10^{3} 4.184*
kilogram mass	kilogram	10^{0} 1.00*
kilogram force (kgf)	newton	10^{0} 9.806 65*
kilopond force	newton	10^{0} 9.806 65*
kip	newton	10^{3} 4 448 221 615 260 5*
knot (international)	meter/second	10^{-1} 5.144 444 444
lambert	candela/meter²	10^{4} 1/π*
lambert	candela/meter²	10^{3} 3.183 098 8
langley	joule/meter²	10^{4} 4.184*
lbf (pound force, avoirdupois)	newton	10^{0} 4 448 221 615 260 5*
lbm (pound mass, avoirdupois)	kilogram	10^{-1} 4.535 923 7*
league (British nautical)	meter	10^{3} 5 556 552*
league (international nautical)	meter	10^{3} 5.556*
league (statute)	meter	10^{3} 4.828 032*
light year	meter	10^{15} 9.460 55
link (engineer or ramden)	meter	10^{-1} 3.04*
link (surveyor or gunter)	meter	10^{-1} 2 011 68*
liter	meter³	10^{-3} 1.00*
lux	lumen/meter²	10^{0} 1.00*
maxwell	weber	10^{-8} 1.00*

Conversion Factors (Cont)

To convert from	to	multiply by
[illegible]	wavelengths Kr 86	+06 1 650 763 73*
[illegible]	meter	−06 1.00*
[illegible]	meter	−05 2.54*
[illegible] (U.S. statute)	meter	+03 1.609 344*
[illegible] (U.K. nautical)	meter	+03 1.853 184*
[illegible] (international nautical)	meter	+03 1.852*
[illegible] (U.S. nautical)	meter	+03 1.852*
[illegible]	newton/meter²	+02 1.00*
[illegible] meter of mercury (0° C)	newton/meter²	+02 1.333 224
[illegible] angle)	radian	−04 2.908 882 086 66
[illegible] (mean solar)	second (mean solar)	+01 6.00*
[illegible] (sidereal)	second (mean solar)	+01 5.983 617 4
[illegible] (mean calendar)	second (mean solar)	+06 2.628*
nautical mile (international)	meter	+03 1.852*
nautical mile (U.S.)	meter	+03 1.852*
nautical mile (U.K.)	meter	+03 1.853 184*
oersted	ampere/meter	+01 7.957 747 2
ohm (international of 1948)	ohm	+00 1.000 495
ounce force (avoirdupois)	newton	−01 2.780 138 5
ounce mass (avoirdupois)	kilogram	−02 2.834 952 312 5*
ounce mass (troy or apothecary)	kilogram	−02 3.110 347 6**
ounce (U.S. fluid)	meter³	−05 2.957 352 956 25*
pace	meter	−01 7.62*
parsec	meter	+16 3.083 74
pascal	newton/meter²	+00 1.00*
peck (U.S.)	meter³	−03 8.809 767 541 72*
pennyweight	kilogram	−03 1.555 173 84*
perch	meter	+00 5.0292*
phot	lumen/meter²	+04 1.00
pica (printers)	meter	−03 4.217 517 6*
pint (U.S. dry)	meter³	−04 5.506 104 713 575*
pint (U.S. liquid)	meter³	−04 4.731 764 75*
point (printers)	meter	−04 3.514 598*
poise	newton second/meter²	−01 1.00*
pole	meter	+00 5.0292*
pound force (lbf avoirdupois)	newton	+00 4.448 221 615 260 5*
pound mass (lbm avoirdupois)	kilogram	−01 4.535 923 7*
pound mass (troy or apothecary)	kilogram	−01 3.732 417 216*
poundal	newton	−01 1.382 549 543 76*
quart (U.S. dry)	meter³	−03 1.101 220 942 715*
quart (U.S. liquid)	meter³	−04 9.463 529 5
rad (radiation dose absorbed)	joule/kilogram	−02 1.00*
Rankine (temperature)	kelvin	$t_K = (5/9)t_R$
rayleigh (rate of photon emission)	1/second meter²	+10 1.00*
rhe	meter²/newton second	+01 1.00*
rod	meter	+00 5.0292*
roentgen	coulomb/kilogram	−04 2.579 76*
rutherford	disintegration/second	+06 1.00*
second (angle)	radian	−06 4.848 136 811
second (*ephemeris*)	second	+00 1.000 000 000
second (mean solar)	second (ephemeris)	Consult American Ephemeris and Nautical Almanac

To convert from	to	multiply by
second (sidereal)	second (mean solar)	−01 9.972 695 7
section	meter²	+06 2.589 988 110 336*
scruple (apothecary)	kilogram	−03 1.295 978 2*
shake	second	−08 1.00
skein	meter	+02 1.097 28*
slug	kilogram	+01 1.459 390 29
span	meter	−01 2.286*
statampere	ampere	−10 3.335 640
statcoulomb	coulomb	−10 3.335 640
statfarad	farad	−12 1.112 650
stathenry	henry	+11 8.987 554
statmho	mho	−12 1.112 650
statohm	ohm	+11 8.987 554
statute mile (U.S.)	meter	+03 1.609 344*
statvolt	volt	+02 2.997 925
stere	meter³	+00 1.00*
stilb	candela/meter²	+04 1.00
stoke	meter²/second	−04 1.00*
tablespoon	meter³	−05 1.478 676 478 125*
teaspoon	meter³	−06 4.928 921 593 75*
ton (assay)	kilogram	−02 2.916 666 6
ton (long)	kilogram	+03 1.016 046 908 8*
ton (metric)	kilogram	+03 1.00*
ton (nuclear equivalent of TNT)	joule	+09 4.20
ton (register)	meter³	+00 2.831 684 659 2*
ton (short, 2000 pound)	kilogram	+02 9.071 847 4*
tonne	kilogram	+03 1.00*
torr (0° C)	newton/meter²	+02 1.333 22
township	meter²	+07 9.323 957 2
unit pole	weber	−07 1.256 637
volt (international of 1948)	volt	+00 1.000 330
watt (international of 1948)	watt	+00 1.000 165
yard	meter	−01 9.144*
year (calendar)	second (mean solar)	+07 3.153 6*
year (sidereal)	second (mean solar)	+07 3.155 815 0
year (tropical)	second (mean solar)	+07 3.155 692 6
year 1900, tropical, Jan., day 0, hour 12	second (ephemeris)	+07 3.155 692 597 47*
year 1900, tropical, Jan., day 0, hour 12	second	+07 3.155 692 597 47

Listing by Physical Quantity

ACCELERATION

foot/second²	meter/second²	−01 3.048*
free fall, standard	meter/second²	+00 9.806 65*
gal (galileo)	meter/second²	−02 1.00*
inch/second²	meter/second²	−02 2.54*

AREA

acre	meter²	+03 4.046 856 422 4*
are	meter²	+02 1.00*
barn	meter²	−28 1.00*
circular mil	meter²	−10 5.067 074 8

To convert from	to	multiply by
foot²	meter²	−02 9.290 304*
hectare	meter²	+04 1.00*
inch²	meter²	−04 6.4516*
mile² (U.S. statute)	meter²	+06 2.589 988 110 336*
section	meter²	+06 2.589 988 110 336*
township	meter²	+07 9.323 957 2
yard²	meter²	−01 8 361 273 6*

DENSITY

gram/centimeter³	kilogram/meter³	+03 1.00*
Lm/Inch³	kilogram/meter³	+04 2.767 990 5
Lm/Foot³	kilogram/meter³	+01 1.601 846 3
slug/foot³	kilogram/meter³	+02 5.153 79

ENERGY

British thermal unit (ISO/TC 12)	joule	+03 1.055 06
British thermal unit (International Steam Table)	joule	+03 1.055 04
British thermal unit (mean)	joule	+03 1 055 87
British thermal unit (thermochemical)	joule	+03 1.054 350 264 488
British thermal unit (39° F)	joule	+03 1.059 67
British thermal unit (60° F)	joule	+03 1.054 68
calorie (International Steam Table)	joule	+00 4.1868
calorie (mean)	joule	+00 4.190 02
calorie (thermochemical)	joule	+00 4.184*
calorie (15° C)	joule	+00 4.185 80
calorie (20° C)	joule	+00 4.181 90
calorie (kilogram, International Steam Table)	joule	+03 4.1868
calorie (kilogram, mean)	joule	+03 4.190 02
calorie (kilogram, thermochemical)	joule	+03 4.184*
electron volt	joule	−19 1.602 10
erg	joule	−07 1.0*
foot lbf	joule	+00 1.355 817 9
foot poundal	joule	−02 4.214 011 0
joule (international of 1948)	joule	+00 1.000 165
kilocalorie (International Steam Table)	joule	+03 4.1868
kilocalorie (mean)	joule	+03 4.190 02
kilocalorie (thermochemical)	joule	+03 4.184*
kilowatt hour	joule	+06 3.60*
kilowatt hour (international of 1948)	joule	+06 3.600 59
ton (nuclear equivalent of TNT)	joule	+09 4.20
watt hour	joule	+03 3.60*

ENERGY/AREA TIME

Btu (thermochemical)/foot² second	watt/meter²	+04 1.134 893 7
Btu (thermochemical)/foot² minute	watt/meter²	+02 1.891 488 5
Btu (thermochemical)/foot² hour	watt/meter²	+00 3.152 480 8
Btu (thermochemical)/inch² second	watt/meter²	+06 1.634 246 2
calorie (thermochemical)/cm² minute	watt/meter²	+02 6.973 333 3
erg centimeter² second	watt/meter²	−03 1.00*
watt/centimeter²	watt/meter²	+04 1.00*

FORCE

dyne	newton	−05 1.00*
kilogram force (kgf)	newton	+00 9.806 65*

Conversion Factors (Cont)

To convert from	*to*	*multiply by*
kilopond force	newton	+00 9.806 65*
kip	newton	+03 4.448 221 615 260 5*
lbf (pound force, avoirdupois)	newton	+00 4.448 221 615 260 5*
ounce force (avoirdupois)	newton	−01 2.780 138 5
pound force, lbf (avoirdupois)	newton	+00 4.448 221 615 260 5*
poundal	newton	−01 1.382 549 543 76*

LENGTH

angstrom	meter	−10 1.00*
astronomical unit	meter	+11 1.495 978 9
cable	meter	+02 2.194 56*
caliber	meter	−04 2.54*
chain (surveyor or gunter)	meter	+01 2 011 68*
chain (engineer or ramden)	meter	+01 3.048*
cubit	meter	−01 4.572*
fathom	meter	+00 1.8288*
fermi (femtometer)	meter	−15 1.00*
foot	meter	−01 3.048*
foot (U.S. survey)	meter	+00 1200 3937*
foot (U.S. survey)	meter	−01 3.048 006 096
furlong	meter	+02 2.011 68*
hand	meter	−01 1.016*
inch	meter	−02 2.54*
league (U.K. nautical)	meter	+05 5 559 552*
league (international nautical)	meter	+03 5.556*
league (statute)	meter	+03 4 828 032*
light year	meter	+15 9.460 55
link (engineer or ramden)	meter	−01 3.048*
link (surveyor or gunter)	meter	−01 2.011 68*
meter	wavelengths Kr 86	+06 1.650 763 73*
micron	meter	−06 1.00*
mil	meter	−05 2.54*
mile (U.S. statute)	meter	+03 1.609 344*
mile (U.K. nautical)	meter	+03 1.853 184*
mile (international nautical)	meter	+03 1.852*
mile (U.S. nautical)	meter	+03 1.852*
nautical mile (U.K.)	meter	+03 1.853 184*
nautical mile (international)	meter	+03 1.852*
nautical mile (U.S.)	meter	+03 1.852*
pace	meter	−01 7.62*
parsec	meter	+16 3.083 74
perch	meter	+00 5.0292*
pica (printers)	meter	−03 4.217 517 6*
point (printers)	meter	−04 3.514 598*
pole	meter	+00 5.0292*
rod	meter	+00 5.0292*
skein	meter	+02 1.097 28*
span	meter	−01 2.286*
statute mile (U.S.)	meter	+03 1.609 344*
yard	meter	−01 9.144*

MASS

carat (metric)	kilogram	−04 2.00*
dram (avoirdupois)	kilogram	−03 1.771 845 195 312 5*
dram (troy or apothecary)	kilogram	−03 3.887 934 6*
grain	kilogram	−05 6.479 891*
gram	kilogram	−03 1.00*

Conversion Factors (Cont)

To convert from	*to*	*multiply by*
hundredweight (long)	kilogram	+01 5.080 234 544*
hundredweight (short)	kilogram	+01 4.535 923 7*
kgf second² meter (mass)	kilogram	+00 9.806 65*
kilogram mass	kilogram	+00 1.0*
lbm (pound mass, avoirdupois)	kilogram	−01 4.535 923 7*
ounce mass (avoirdupois)	kilogram	−02 2.834 952 312 5*
ounce mass (troy or apothecary)	kilogram	−02 3.110 347 68*
pennyweight	kilogram	−03 1.555 173 84*
pound mass, lbm (avoirdupois)	kilogram	−01 4.535 923 7*
pound mass (troy or apothecary)	kilogram	−01 3.732 417 216*
scruple (apothecary)	kilogram	−03 1.295 978 2*
slug	kilogram	+01 1.459 390 29
ton (assay)	kilogram	−02 2.916 666 6
ton (long)	kilogram	+03 1.016 046 908 8*
ton (metric)	kilogram	+03 1.00*
ton (short, 2000 pound)	kilogram	+02 9.071 847 4*
tonne	kilogram	+03 1.00*

POWER

Btu (thermochemical)/second	watt	+03 1.054 350 264 488
Btu (thermochemical)/minute	watt	+01 1.757 250 4
calorie (thermochemical)/second	watt	+00 4.184*
calorie (thermochemical)/minute	watt	−02 6.973 333 3
foot lbf hour	watt	−04 3.766 161 0
foot lbf minute	watt	−02 2.259 696 6
foot lbf second	watt	+00 1.355 817 9
horsepower (550 foot lbf/second)	watt	+02 7.456 998 7
horsepower (boiler)	watt	+03 9.809 50
horsepower (electric)	watt	+02 7.46*
horsepower (metric)	watt	+02 7.354 99
horsepower (U.K.)	watt	+02 7.457
horsepower (water)	watt	+02 7.460 43
kilocalorie (thermochemical)/minute	watt	+01 6.973 333 3
kilocalorie (thermochemical)/second	watt	+03 4.184*
watt (international of 1948)	watt	+00 1.000 165

PRESSURE

atmosphere	newton/meter²	+05 1.013 25*
bar	newton/meter²	+05 1.00*
barye	newton/meter²	−01 1.00*
centimeter of mercury (0° C)	newton/meter²	+03 1.333 22
centimeter of water (4° C)	newton/meter²	+01 9.806 38
dyne/centimeter²	newton/meter²	−01 1.00*
foot of water (39.2° F)	newton/meter²	+03 2.988 98
inch of mercury (32° F)	newton/meter²	+03 3.386 389
inch of mercury (60° F)	newton/meter²	+03 3 376 85
inch of water (39.2° F)	newton/meter²	+02 2.490 82
inch of water (60° F)	newton/meter²	+02 2.4884
kgf/centimeter²	newton/meter²	+04 9.806 65*
kgf/meter²	newton/meter²	+00 9 806 65*
lbf/foot²	newton/meter²	+01 4.788 025 8
lbf/inch² (psi)	newton/meter²	+03 6.894 757 2
millibar	newton/meter²	+02 1.00*
millimeter of mercury (0° C)	newton/meter²	+02 1.333 224
pascal	newton/meter²	+00 1.00*
psi (lbf/inch²)	newton/meter²	+03 6.894 757 2
torr (0° C)	newton/meter²	+02 1.333 22

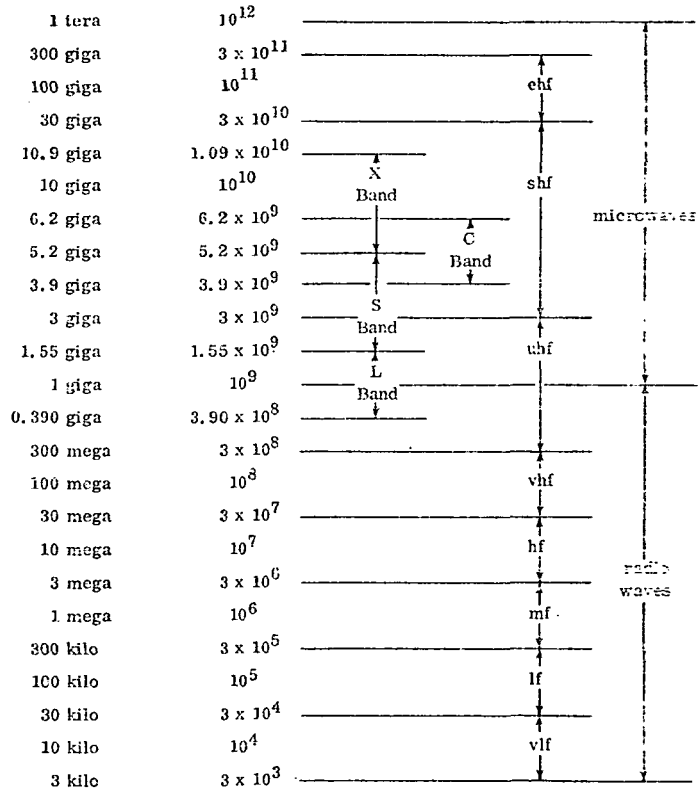
Conversion Factors (Cont)

To convert from	to	multiply by
SPEED		
foot/hour	meter/second	−05 8.466 666 5
foot/minute	meter/second	−03 5.08*
foot/second	meter/second	−01 3.048*
inch/second	meter/second	−02 2.54*
kilometer/hour	meter/second	−01 2.777 777 8
knot (international)	meter/second	−01 5.144 444 444
mile/hour (U.S. statute)	meter/second	−01 4.4704*
mile/minute (U.S. statute)	meter/second	+01 2.682 24*
mile/second (U.S. statute)	meter/second	+03 1.609 344*
TEMPERATURE		
Celsius	kelvin	$t_K = t_C + 273.15$
Fahrenheit	kelvin	$t_K = (5/9)(t_F + 459.67)$
Fahrenheit	Celsius	$t_C = (5/9)(t_F - 32)$
Rankine	kelvin	$t_K = (5/9)t_R$
TIME		
day (mean solar)	second (mean solar)	+04 8.64*
day (sidereal)	second (mean solar)	+04 8 616 409 v
hour (mean solar)	second (mean solar)	+03 3 6[illegible]*
hour (sidereal)	second (mean solar)	+03 3 590 170 4
minute (mean solar)	second (mean solar)	+01 6.0*
minute (sidereal)	second (mean solar)	+01 5 983 617 4
month (mean calendar)	second (mean solar)	+06 2 628*
second (ephemeris)	second	+00 1.000 000 000
second (mean solar)	second (ephemeris)	Consult American Ephemeris and Nautical Almanac
second (sidereal)	second (mean solar)	−01 9 972 695 7
year (calendar)	second (mean solar)	+07 3.1536*
year (sidereal)	second (mean solar)	+07 3.155 815 0
year (tropical)	second (mean solar)	+07 3.155 692 6
year 1900, tropical, Jan., day 0, hour 12	second (ephemeris)	+07 3.155 692 597 47*
year 1900, tropical, Jan., day 0, hour 12	second	+07 3.155 692 597 47
VISCOSITY		
centistoke	meter²/second	−06 1.00*
stoke	meter²/second	−01 1.00*
foot²/second	meter²/second	−02 9.290 304*
centipoise	newton second/meter²	−03 1.00*
lbm/foot second	newton second/meter²	+00 1.488 163 9
lbf second/foot²	newton second/meter²	+01 4.788 025 8
poise	newton second/meter²	−01 1.00*
poundal second/foot²	newton second/meter²	+00 1.488 163 9
slug/foot second	newton second/meter²	+01 4.788 025 8
rhe	meter²/newton second	+01 1.00*
VOLUME		
acre foot	meter³	+03 1.233 481 9
barrel (petroleum, 42 gallons)	meter³	−01 1.589 873
board foot	meter³	−03 2.359 737 216*
bushel (U.S.)	meter³	−02 3.523 907 016 688*
cord	meter³	+00 3.624 556 3

Conversion Factors (Cont)

To convert from	*to*	*multiply by*
cup	meter³	−04 2.365 882 365*
dram (U.S. fluid)	meter³	−06 3.696 691 195 312 5*
fluid ounce (U.S.)	meter³	−05 2.957 352 956 25*
foot³	meter³	−02 2.831 684 659 2*
gallon (U.K. liquid)	meter³	−03 4.546 087
gallon (U.S. dry)	meter³	−03 4.404 883 770 86*
gallon (U.S. liquid)	meter³	−03 3.785 411 784*
gill (U.K.)	meter³	−04 1.420 652
gill (U.S.)	meter³	−04 1.182 941 2
hogshead (U.S.)	meter³	−01 2.384 809 423 92*
inch³	meter³	−05 1.638 706 4*
liter	meter³	−03 1.00*
ounce (U.S. fluid)	meter³	−05 2.957 352 956 25*
peck (U.S.)	meter³	−03 8.809 767 541 72*
pint (U.S. dry)	meter³	−04 5.506 104 713 575*
pint (U.S. liquid)	meter³	−04 4.731 764 73*
quart (U.S. dry)	meter³	−03 1.101 220 942 715*
quart (U.S. liquid)	meter³	−04 9 463 529 5
stere	meter³	+00 1.00*
tablespoon	meter³	−05 1.478 676 478 125*
teaspoon	meter³	−06 4.928 921 593 75*
ton (register)	meter³	+00 2.831 684 659 2*
yard³	meter³	−01 7.645 548 579 84*

Radio Frequency Spectrum Pertinent to ATS-F and -G



Instructions for Determining Antenna Azimuth and Elevation Angles to any Equatorial Synchronous Satellite

1. From the known station and subsatellite coordinates, determine distance from subsatellite to ground station in degrees of latitude (Δ latitude) and degrees of longitude (Δ longitude).

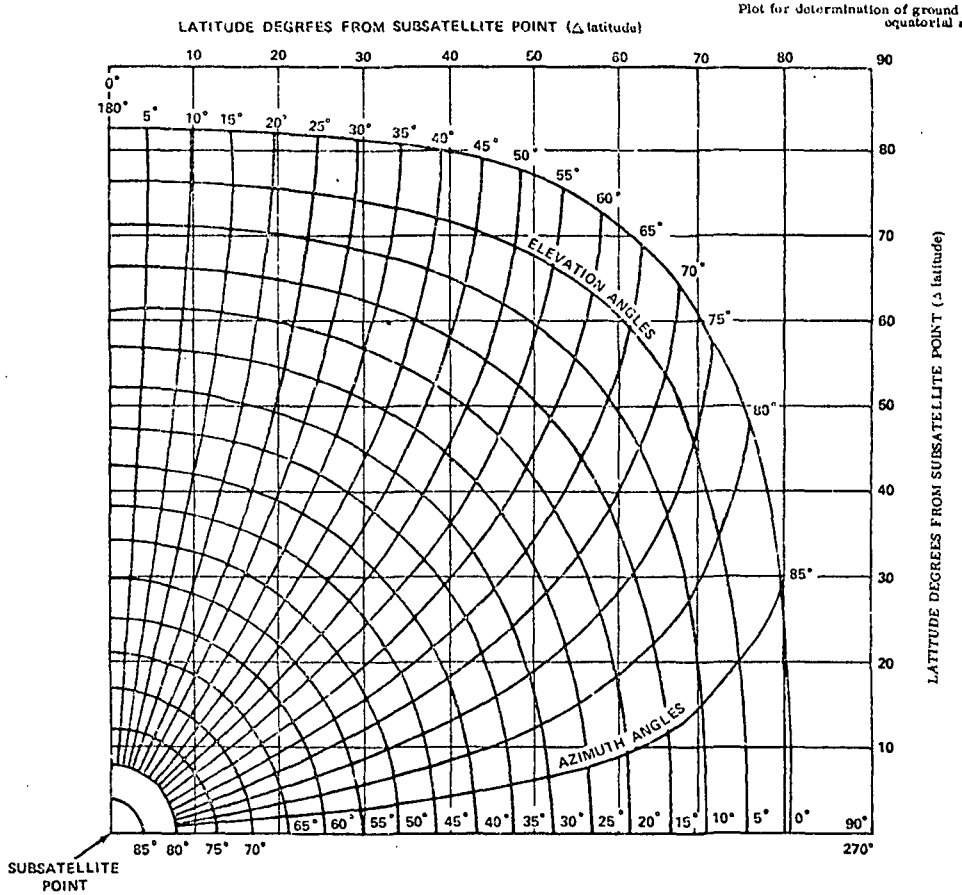
2. On the graph locate the point determined by Δ longitude and Δ latitude.

3. From the graph determine the elevation and nominal azimuth angles.

4. From the geography determine in which quadrant the satellite lies with respect to the ground station (imagine the small "quadrant angle" plot superimposed over the ground station).

5. From the table, convert nominal azimuth to actual azimuth.

Plot for determination of ground station azimuth and elevation angles to any equatorial synchronous satellite.


LATITUDE DEGREES FROM SUBSATELLITE POINT (Δ latitude)
10 20 30 40 50 60 70 80 90
0° 180° 5° 10° 15° 20° 25° 30° 35° 40° 45° 50° 55° 60° 65° 70° 75° 80° 85°
ELEVATION ANGLES
AZIMUTH ANGLES
65° 60° 55° 50° 45° 40° 35° 30° 25° 20° 15° 10° 5° 0° 90° 270°
85° 80° 75° 70°
SUBSATELLITE POINT
LATITUDE DEGREES FROM SUBSATELLITE POINT (Δ latitude)
80 70 60 50 40 30 20 10

Quadrant Angles
(Ground Station at Center)

N
0°
W 270° Q4 | Q1 90° E
Q3 | Q2
180°
S

AZIMUTH ANGLE CONVERSION
(Q1 ANGLES ARE SHOWN ON CHART)

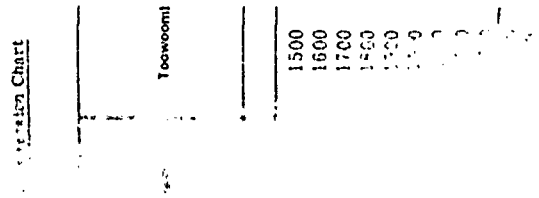
Q1	Q2	Q3	Q4
0	180	180	0
5	175	185	355
10	170	190	350
15	165	195	345
20	160	200	340
25	155	205	335
30	150	210	330
35	145	215	325
40	140	220	320
45	135	225	315
50	130	230	310
55	125	235	305
60	120	240	300
65	115	245	295
70	110	250	290
75	105	255	285
80	100	260	280
85	95	265	275
90	90	270	270

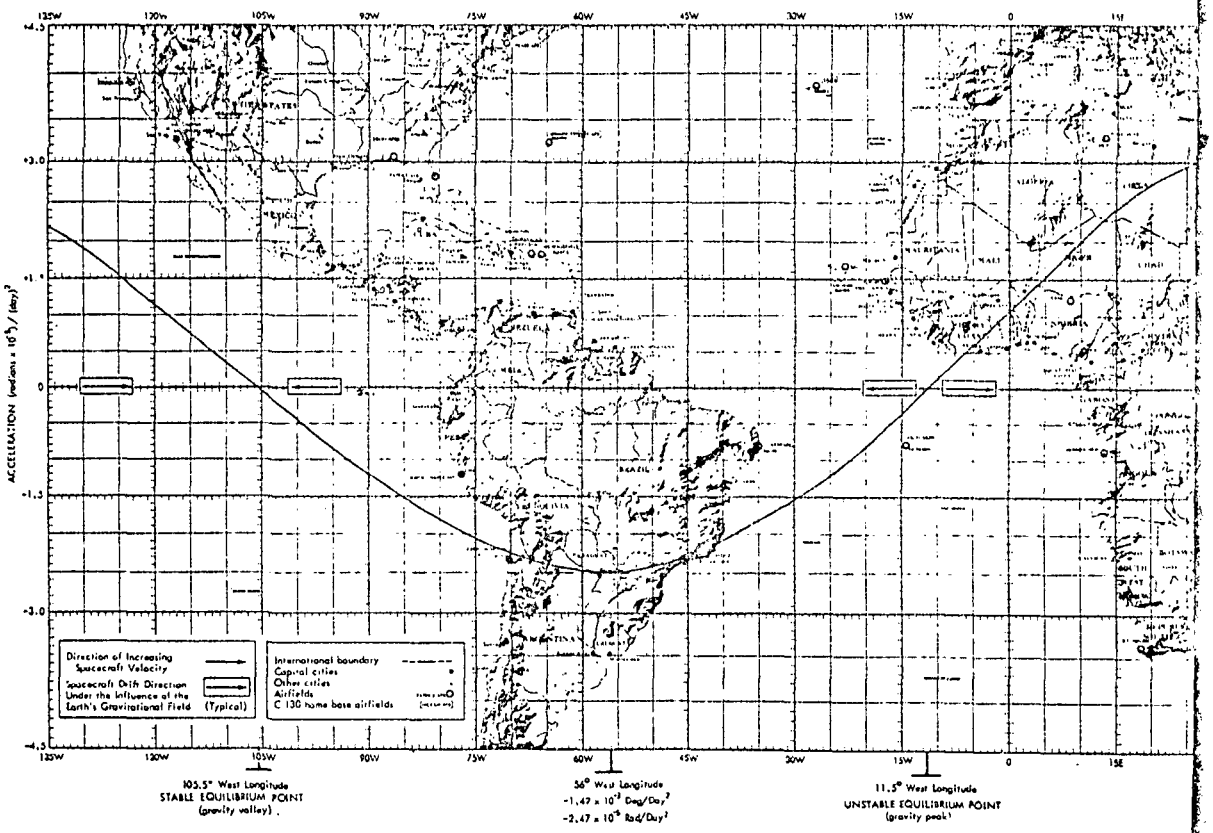
Time Conversion Chart

	Mojave		Rosman GSFC		Rio	England Spain	Pleumeur Bodou Fuchino Raisting Rao	Ahmedabad	Japan	Toowoomba	ATS-1 Sub-Satellite Point	
	PST	PDT MST	EST	EDT AST		GMT						
Yesterday	2100	2200	0000	0100	0200	0500	0600	1030	1400	1500	1600	Today
	2200	2300	0100	0200	0300	0600	0700	1130	1500	1600	2000	
	2300	0000	0200	0300	0400	0700	0800	1230	1600	1700	2100	
Today	0000	0100	0300	0400	0500	0800	0900	1330	1700	1800	2200	
	0100	0200	0400	0500	0600	0900	1000	1430	1800	1900	2300	
	0200	0300	0500	0600	0700	1000	1100	1530	1900	2000	0000	Tomorrow
	0300	0400	0600	0700	0800	1100	1200	1630	2000	2100	0100	
	0400	0500	0700	0800	0900	1200	1300	1730	2100	2200	0200	
	0500	0600	0800	0900	1000	1300	1400	1830	2200	2300	0300	
	0600	0700	0900	1000	1100	1400	1500	1930	2300	0000	0400	
	0700	0800	1000	1100	1200	1500	1600	2030	0000	0100	0500	
	0800	0900	1100	1200	1300	1600	1700	2130	0100	0200	0600	
	0900	1000	1200	1300	1400	1700	1800	2230	0200	0300	0700	
	1000	1100	1300	1400	1500	1800	1900	2330	0300	0400	0800	
	1100	1200	1400	1500	1600	1900	2000	0030	0400	0500	0900	
	1200	1300	1500	1600	1700	2000	2100	0130	0500	0600	1000	
	1300	1400	1600	1700	1800	2100	2200	0230	0600	0700	1100	
	1400	1500	1700	1800	1900	2200	2300	0330	0700	0800	1200	
	1500	1600	1800	1900	2000	2300	0000	0430	0800	0900	1300	
	1600	1700	1900	2000	2100	0000	0100	0530	0900	1000	1400	
	1700	1800	2000	2100	2200	0100	0200	0630	1000	1100	1500	
	1800	1900	2100	2200	2300	0200	0300	0730	1100	1200	1600	
	1900	2000	2200	2300	0000	0300	0400	0830	1200	1300	1700	
	2000	2100	2300	0000	0100	0400	0500	0930	1300	1400	1800	

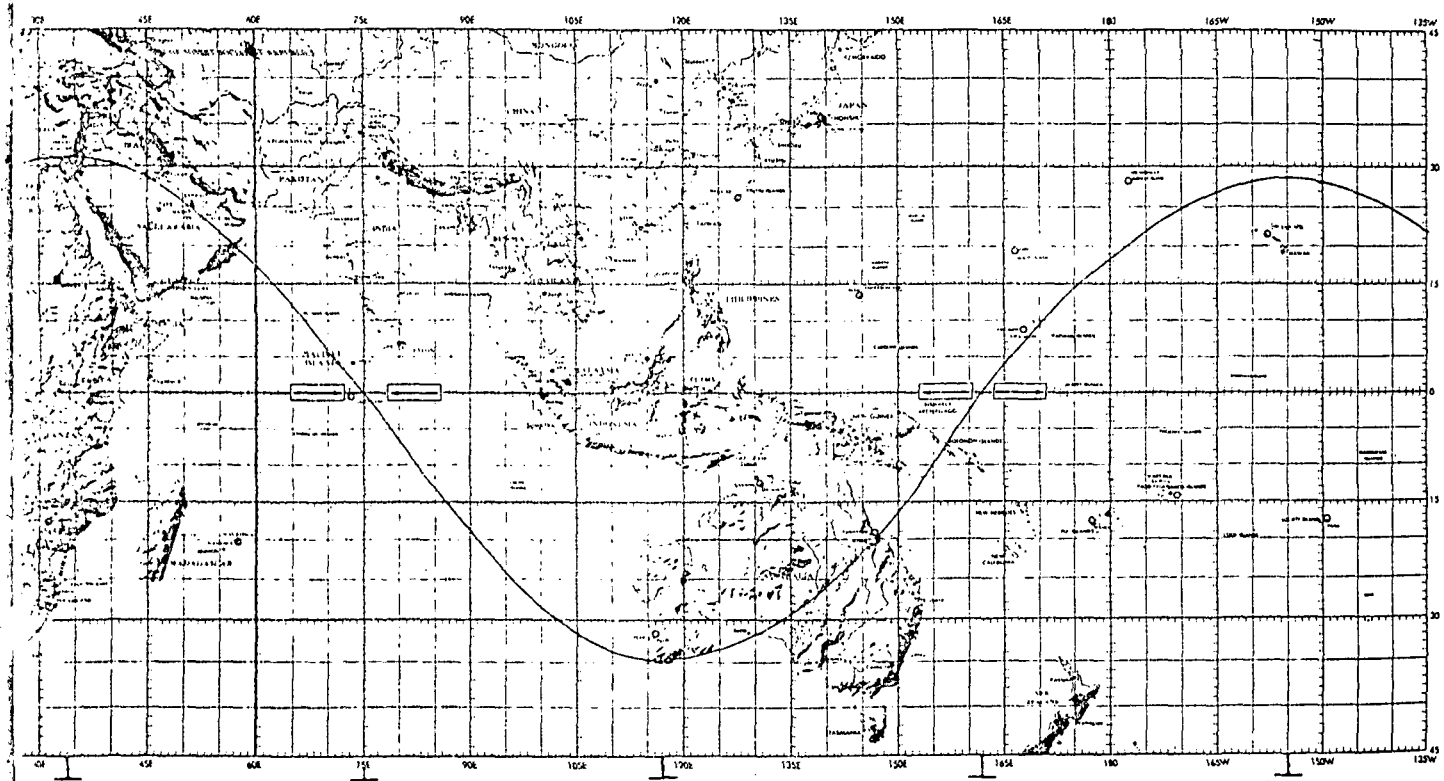
Elevation Angles for Visibility with a Satellite at Synchronous Altitude,
0 Latitude and Longitudes Shown; from the Listed Stations

STATION			LONGITUDE												LONGITUDE												STATION		
			WEST												EAST														
			180°	165°	150°	135°	120°	105°	90°	75°	60°	45°	30°	15°	0°	15°	30°	45°	60°	75°	90°	105°	120°	135°	150°	165°			
Rosman	35°11'35.4"	N																									Rosman	35°11'35.4"	N
	82°52'22"	W			9.96	22.1	33.4	42.9	48.4	48.3	42.5	32.9	21.5	9.3														82°52'22"	W
Mojave	35°17'48"	N																									Mojave	35°17'48"	N
	116°53'57"	W	13.2	25.2	36.1	44.7	48.9	47.1	40.1	24.8	18.2	6.1														1.0		116°53'57"	W
Toowoomba	27°23'	S																									Toowoomba	27°23'	S
	151°57'	E	45.5	33.2	19.8	6.4														2.9	16.3	29.8	42.5	52.9	58.0	54.9		151°57'	E
Japan	36°41'39.726"	N																									Japan	36°41'39.726"	N
(KDD)	140°41'42.598"	E	30.9	19.7	7.8															10.7	22.5	33.4	42.2	47.0	46.3	40.4	(KDD)	140°41'42.598"	E
Japan	35°57'10.0"	N																									Japan	35°57'10.0"	N
(RRL)	140°39'57.8"	E	31.4	20.0	8.0															11.0	22.9	33.9	42.9	47.9	47.1	41.0	(RRL)	140°39'57.8"	E
India	23°	N																									India	23°	N
Ahmedabad	73°	E													7.0	21.1	35.2	48.6	59.4	63.0	56.9	45.2	31.5	17.3	3.3		Ahmedabad	73°	E
Italy	41°58'41"	N																									Italy	41°58'41"	N
Fuchino	13°36'05"	E									3.4	14.4	24.7	33.5	39.6	41.5	38.7	32.0	22.8	12.4	1.4						Fuchino	13°36'05"	E
Sweden	57°23'36.1"	N																									Sweden	57°23'36.1"	N
Rao	11°55'12.8"	E									1.0	8.5	15.3	20.6	23.9	24.7	22.8	18.6	12.6	5.5							Rao	11°55'12.8"	E
Germany	47°54'08.58"	N																									Germany	47°54'08.58"	N
Raisting	11°06'59.12"	E									3.9	13.5	22.3	29.5	34.0	34.9	32.0	26.0	17.9	8.6							Raisting	11°06'59.12"	E
France	48°47'13"	N																									France	48°47'13"	N
Pleumeur Bodou	3°31'20.5"	W								3.4	12.9	21.5	28.5	32.9	33.9	31.3	25.5	17.6	8.5								Pleumeur Bodou	3°31'20.5"	W
England	50°02'58"	N																									England	50°02'58"	N
Goonhilly	5°10'29"	W								4.1	13.3	21.5	28.0	31.9	32.4	29.5	23.7	15.9	7.0								Goonhilly	5°10'29"	W
Canada	44°11'24"	N																									Canada	44°11'24"	N
Mill Village	64°40'00"	W				5.3	15.7	25.3	33.1	38.1	38.9	35.4	28.5	19.5	9.3												Mill Village	64°40'00"	W




ACCELERATION (radians x 10⁵) / (day)²
Direction of Increasing Spacecraft Velocity
Spacecraft Drift Direction Under the Influence of the Earth's Gravitational Field (Typical)
International boundary
Capital cities
Other cities
Airfields
C 130 home base airfields
105.5° West Longitude
STABLE EQUILIBRIUM POINT
(gravity valley)
56° West Longitude
-1.47 x 10⁻³ Deg/Day²
-2.47 x 10⁻⁵ Rad/Day²
11.5° West Longitude
UNSTABLE EQUILIBRIUM POINT
(gravity peak)

...TE LONGITUDINAL DRIFT ACCELERATION VS LONGITUDE



...° East Longitude
...° x 10^{-2} Deg/Day^2
...4 x 10^{-4} Rad/Day^2

...2.0° East Longitude
STABLE EQUILIBRIUM POINT
(gravity valley)

117° East Longitude
-1.98 x 10^{-3} Deg/Day^2
-3.45 x 10^{-5} Rad/Day^2

161.8° East Longitude
UNSTABLE EQUILIBRIUM POINT
(gravity peak)

155° West Longitude
1.64 x 10^{-3} Deg/Day^2
2.86 x 10^{-5} Rad/Day^2

Prepared by C. D. Bullock, Code 460 per information from Carl A. Wagner, Code 552, 7/31/70

Transmission Links Data

USER	LINK	FREQ. MHz	BW (MHz)	XMIT. PWR WATTS	EIRP DBW	ANTENNA TRANSMIT TYPE	TRANSMIT FOV (DEGREES)	TRANSMIT PEAK GAIN (db)	TRANSMIT FOV GAIN (db)	TRANSMIT POLARIZATION	ANTENNA RECEIVE TYPE	RECEIVE FOV (DEGREES)	RECEIVE PEAK GAIN (db)	RECEIVE FOV GAIN (db)	*G/Ts (db/°K)	FREQUENCY ALLOCATION REQUESTED	FREQUENCY ALLOCATION RECEIVED	REMARKS
PLACE	GND TERM TO S/C	1,660	2.5	1,000	62.3	15°P	3	34.3	31.3	RCP	30°P	7.5/11.0 1.5/1.3 1.26	31.5 38.5 +9	28.5 36.5 +4.5	-5.0 (FOV) 5.5 -26.0 (FOV)	1,660	YES	S/C L BAND AND C BAND ANTENNA REQUIRED SIMULTANEOUSLY
		5,950 6,150 6,350	2	8,000 8,000 2,000	97.0 90.0 79.0	85°P 40°P 21°P	0.14 0.29 0.55	61.0 53.0 48.9	58.0 50.0 45.9	LINEAR	HORN	20	16.6	13.6	20.0 (FOV)	5,950 6,150 6,350	YES	
	S/C TO AIRCRAFT	1,550	4	40	46 51	30°P	7.5/11.0 1.6	31.5 38.5	28.5 35.5	RCP	EXP	120.54	3.10	0.7	-26.4 (FOV)	1,550	YES	
	AIRCRAFT TO S/C	1,650	2.5	250	25.3	EXP.	120.54	3.10	0.7	RCP	30°P	7.5/11.0 1.3	31.5 38.5	28.5 35.5	- 8.0 (FOV) 5.5	1,650	YES	
	S/C TO GND TERM.	1550 3,750 3,950 4,150	2.5	40	>23.5	30°P	>20	>9	>4.5	RCP	15°P 85°P 40°P 21°P	3 0.21 0.46 0.87	31.6 56.0 49.5 44.8	30.6 53.0 46.5 41.8	10.6 36.2 31.6 24.8	1,550 3,750 3,950 4,150	YES	
DATA RELAY EXPERIMENT (DRE)	GND TERM TO S/C	5,950 6,150 6,350	0.12	8,000 8,000 2,000	98.0 90.0 79.0	85°P 40°P 21°P	0.14 0.29 0.55	61.0 53.0 48.9	58.0 50.0 45.9	LINEAR	HORN	-20 FOV	16.6	-13.6	-20 FV	5,950 6,150 6,350	YES	
		2,253	40/12	130	54.9	15°P	2.2	38.9	33.9	RCP	30°P	1.3	40.4	36.6	9.6	2,247 2,253	YES	MONOPULSE POINT
	S/C TO GND TERM	3,750 3,950 4,150	40	20	28	HORN	20	16.7	13.7	LINEAR	85°P 40°P 21°P	0.21 0.47 0.87	56.0 49.5 44.8	53.0 46.5 41.8	36.2 30.5 24.8	3750 3,950 4,150	YES	
		2,075	40	20	49.2	30°P	1.2	40.7	37.6	RCP	15°P	1.4	34.8	31.8	10.6	2,075	NO	
	NIMBUS TO S/C	2,253	40/12	2, 4, 8	NA	QUAD HELIX	26	18	13	RCP	30°P	1.3	40.4	36.6	7(SCAN)	2,253	YES	
	S/C TO NIMBUS	2,075 (2062.26)	0.5	20	49.2 SCAN	30°P	1.2	40.7	37.6	RCP	QUAD HELIX	26	31.6	12	NA	2,075	NO	
MILLIMETER WAVE (GSFC)	S/C TO GND TERM	20,000	0.0001	2	30	HORN	5°x27°	27	24	LINEAR	15°P	0.25	55.0	52.0	20.9	20 (XX)	YES	CW PROPOGATION MODE
		30,000	0.0001	2		HORN	5°x27°	27	24	LINEAR	15°P	0.12	57.4	54.4	28.0	30 (XX)	YES	
	S/C TO GND TERM	20,000	1,440	0.06	14.8	HORN	5°x27°	30	27.5	LINEAR	15°P	0.25	55	52	23.3 (FOV)	20,000	YES	MULTITONE PROPAGATION MODE (per tone 9 tones)
		30,000	1,440	0.06	14.8	HORN	5°x27°	30	27.5	LINEAR	15°P	0.12	57.4	54.4	28.3 (FOV)	30,000	YES	
	S/C TO GND TERM	20,150	40	2	46	15°P	2.4	37	34	LINEAR	15°P	0.25	55.0	52.0	20.9	20,150	YES	COMMUNICATION MODE
		30,150	40	2	42		1.6	39	36	LINEAR	15°P	0.12	57.4	54.4	28.9	30,150	YES	
	S/C TO GND TERM	3,750 3,950 4,150	40	20	28	HORN	20.0	16.7	13.7	LINEAR	85°P	0.21	56	53	36.2	3,750 3,950 4,150	YES	
	GND TERM TO S/C	5,950 6,150 6,350	40	8,000	98.0	85°P	0.14	61.0	58.0	LINEAR	30°P HORN	0.40 20.0	40.4 16.6	46.4 13.6	13.5 17.0	5,950 6,150 6,350	YES	COMMUNICATION MODE
MONOPULSE	GND TERM TO S/C	5,950 6,150 6,350	0.1	8,000 8,000 2,000	98.0 90.0 79.0	85°P 40°P 21°P	0.14 0.29 0.55	61.0 53.0 48.9	58.0 50.0 45.9	LINEAR	30°P	0.40	49.9	46.4	13.5	5,950 6,150 6,350	YES	
		2,253	0.1	100	59.9	15°P	2.2	38.9	33.9	RCP	30°P	1.1	40.4	36.6	9.6	2,253	YES	
	GND TERM TO S/C	154.20 148.26	7.0	2,560	49	SCAMP	24	17	14	RCP	70°P	16	18	15	19	154.20 148.26	YES	

Transmission Links Data

USLR	LINK	FREQ. MHz	BW (MHz)	XMIT. PWR WATTS	EIRP DBW	ANTENNA TRANSMIT TYPE	TRANSMIT FOV (DEGREES)	TRANSMIT PEAK GAIN (db)	TRANSMIT FOV GAIN (db)	TRANSMIT POLARIZATION	ANTENNA RECEIVE TYPE	RECEIVE FOV (DEGREES)	RECEIVE PEAK GAIN (db)	RECEIVE FOV GAIN (db)	*G/Ts (db/ K)	FREQUENCY ALLOCATION REQUESTED	FREQUENCY ALLOCATION RECEIVED	REMARKS
PROPAGATION EXPERIMENT (MNW)	S/C TO GND TERM	4150	10	37 20.9	2.1 12.9	EC HORN	20	17	14	LINEAR	40'P	0.4	52	49	31.5	4,150	YES	
	GND TERM TO S/C	13,100 17,290	10	20	46 48.5	1.58	30 40	37 34.5	34 31.5	LINEAR	1°P	4/8	25.3 28.0	22.3 25.3		13,100 17,790	NO	CW PROP MODE (24 CW TONES)
TRUST	GND TERM TO S/C	5,950 6,150 6,350	10	8,000 8,000 2,000	90.0 90.0 79.9	85°P 40°P 21°P	0.14 0.29 0.56	61.0 53.0 48.9	58.0 50.0 46.9	LINEAR	HORN	20	16.6	13.6	17	5,950 6,150 6,350	YES	ANY TRANSMITTER TO EITHER RECEIVE ANTENNA
		860	40	80	51	30°P	2.8	33	30	LINEAR	30°P	0.4	49.4	46.4	13.5			
										RCP	21°P	4.0	31.7	28.5	5.2	860		NASA FREQ SUPPORT FOR CTR. TO FAB & PRE LAUNCH TESTING REC'D 8/68
	S/C TO GND TERM	3,750 3,950 4,150	40	20	28	HORN	20	16.7	13.7	RCP	10°P	8.0	29.0	21.0	NA			
										LINEAR	85°P	0.21	56.0	53.0	36.7	3,750		
											40°P	0.46	49.5	46.5	31.5	3,950	YES	
											21°P	0.87	44.8	41.8	24.8	4,150		
ITV	GND TERM TO S/C	5,950 6,150 6,350	40	8,000 8,000 2,000	90 90 79.9	40°P 40°P 21°P	0.29 0.29 0.56	56.5 53.0 48.9	53.5 50.0 46.9	LINEAR	HORN 30°P	20	16.6	13.7	20	5,950 6,150 6,350	YES	ANY TRANSMITTER TO EITHER RECEIVE ANTENNA 46°P ANTENNA INDIA
	S/C TO GND TERM	860	40	80	51	30°P	2.8	33	30	HCP	NA	NA	NA	NA	NA	860	NO	INDIAN RECEIVERS
										RCP	21°P	0.87	44.8	41.8	6.2	860	NO	
		3,750 3,950 4,150		20	28	HORN	20	16.7	13.7	LINEAR	40°P	0.46	49.5	46.5	20.5	3,750 3,950 4,150	YES	
R&RR	GND TERM TO S/C	5,950 6,150 6,350	5	8,000 8,000	90.0 90.0	85°P 40°P	0.14 0.29	61.0 53.0	58.0 50.0	LINEAR	HORN	20	16.6	13.6	20(OFV)	5,950 6,150 6,350	YES	
	S/C TO GND TERM	3,750 3,950 4,150	5	20	28	HORN	20.0	16.7	13.7	LINEAR	85°P 40°P 21°P	0.21 0.46 0.87	56.0 49.5 44.8	53.0 46.5 41.8	36.7 31.5 24.8	3,750 3,950 4,150	YES	
INTERFEROMETER	GND TERM TO S/C	6,150 6,155	-	8,000 8,000 2,000	90.0 90.0 79.9	85°P 40°P 21°P	0.14 0.29 0.56	61.0 53.0 48.9	58.0 50.0 46.9	LINEAR	HORN	35	12	10.9	-19.4	6,150 6,155	-YES-	
RFI	GND TERM TO S/C	6175	RFI SOURCES							LIN/SEL	Parabola 30'	0.4	49	46	13.5	N/A	N/A	RFI UP
	GND TERM TO S/C									LINEAR	EC HORN	20	18	15	-20	N/A	N/A	
	GND TERM TO S/C	6175 ± 250	20 TONE 0.001	8	40	6°P	1.0	39	35	LINEAR	HORN	20	18	15	-20	6175	NO	20 CAL. CARRIERS
	GND TERM TO S/C	5,975 6,175 6,375	0.001	8	40	85°P	0.14	60	57	LIN/SEL	30°P	0.4	45	46	+13.5	6,175 ± 250	NO	CONTROL LINK UP
	S/C TO GND TERM	3950	500	20	+11	S/C HORN	20	16.7	13.7	LINEAR	85°P	0.21	56	53	+36	3,950	YES	RFI DATA DOWN
	S/C TO GND TERM	3,750 3,950 4,150	0.001	20	+11	S/C HORN	20	16.7	13.7	LINEAR	6°P	2.8	35	32		3,750 3,650 4,150	YES	CONTROL LINK DOWN

Transmission Links Data

USER	LINK	FREQ. MHz	BW (MHz)	XMIT. PWR WATTS	EIRP DBW	ANTENNA TRANSMIT TYPE	ANTENNA TRANSMIT FOV (DEGREES)	ANTENNA TRANSMIT PEAK GAIN (db)	ANTENNA TRANSMIT FOV GAIN (db)	ANTENNA POLARIZATION	ANTENNA RECEIVE TYPE	ANTENNA RECEIVE FOV (DEGREES)	ANTENNA RECEIVE PEAK GAIN (db)	ANTENNA RECEIVE FOV GAIN (db)	*G/T$_s$ (db/ K)	FREQUENCY ALLOCATION REQUESTED	FREQUENCY ALLOCATION RECEIVED	REMARKS
VERY HIGH RESOLUTION RADIOMETER	S/C TO GND TERM	3,750 3,950 4,150	4	20	23	HORN	20.0	16.7	13.7	LINEAR	85°P 40°P 21°P	0.21 0.46 0.87	56.0 49.5 44.8	53.0 46.5 41.8	36.2 31.5 24.8	3,750 3,950 4,150	YES	
RADIO-BEACON	S/C TO GND TERM	39.0156	~	0.35	-3	DIPOLE	53 E 40 H	2	2	LINEAR					N/A	39.0156	YES	
	S/C TO GND TERM	39.9156	~	0.18	-6	DIPOLE	53 E 40 H	2	2	LINEAR					N/A	39.9156	YES	
	S/C TO GND TERM	40.016	~	0.35	-3	DIPOLE	53 E 40 H	2	2	LINEAR					N/A	40.016	YES	
	S/C TO GND TERM	40.1604	~	0.18	-6	DIPOLE	53 E 40 H	2	2	LINEAR					N/A	40.1604	YES	
	S/C TO GND TERM	41.0164	~	0.35	-3	DIPOLE	53 E 40 H	2	2	LINEAR					N/A	41.0164	YES	
	S/C TO GND TERM	139.0556	~	0.09	-4	MONO	19 E 40 H	2	7	LINEAR					N/A	139.0556	YES	
	S/C TO GND TERM	140.056	~	0.39	+2	MONO	19 E 40 H	7	7	LINEAR					N/A	140.056	YES	
	S/C TO GND TERM	141.0564	~	0.09	-4	MONO	19 E 40 H	7	7	LINEAR					N/A	141.0564	YES	
	S/C TO GND TERM	359.1436	~	0.22	-10	SLOT	116 E 79 H	7	7	LINEAR					N/A	359.1436	YES	
	S/C TO GND TERM	359.04396	~	0.11	-3	SLOT	116 E 40 H	7	7	LINEAR					N/A	359.04396	YES	
	S/C TO GND TERM	360.144	~	0.45	+3	SLOT	116 E 48 H	7	7	LINEAR					N/A	360.144	YES	
	S/C TO GND TERM	360.2440	~	0.11	-3	SLOT	116 E 40 H	7	7	LINEAR						360.2440		
	S/C TO GND TERM	361.1444	~	0.022	-10	SLOT	116 E 40 H	7	7	LINEAR						361.1444		
COMMAND	GND TERM TO S/C	154.20	0.03	2,500	49	ATS SCAMP	24	17	14	CIRCULAR / LINEAR	NEAR OMNI	NEAR OMNI	-3	-10	-44	154.20	YES	PRIME COMMAND
	GND TERM TO S/C	154.20	0.03	2,500	49	ATS SCAMP	24	17	14	RCP	PRIME FOCUS	15°	21	+18	-16	154.20	YES	BACK UP COMMAND
	GND TERM TO S/C	≈6000	12	8,000	95	85°P	0.14	61.0	53.0	LINEAR / LINEAR	HORN	20°	16.6	13.7	-20.4	≈6000	YES	BACK UP COMMAND
					20	40°P	0.29	53.0	50.0									
	GND TERM TO S/C	148.26	0.03	2,500	49	ATS SCAMP	24	17	14	CIRCULAR / LINEAR	NEAR OMNI	NEAR OMNI	-3	-10	-44	148.26	YES	BACK UP COMMAND
	GND TERM TO S/C	148.26	0.03	2,500	49	ATS SCAMP	24	17	14	RCP	PRIME FOCUS	15°	21	+18	-16	148.26	YES	BACK UP COMMAND
TELEMETRY	S/C TO GND TERM	136.23	0.03	5	2	NEAR OMNI	NEAR OMNI	-1	-10	LINEAR / CIRCULAR	ATS SATAN	15	20.3	+17.3	-13.1	136.23	YES	PRIME TELEMETRY
	S/C TO GND TERM	136.23	0.03	5	2	PRIME FOCUS	13°	17	14	RCP	ATS SATAN	15	20.3	+17.3	-13.1	136.23	YES	BACK UP TELEMETRY
	S/C TO GND TERM	137.11	0.03	5	2	NEAR OMNI	NEAR OMNI	-1	-10	LINEAR / CIRCULAR	ATS SATAN	15	20.3	+17.3	-13.1	137.11	YES	PRIME TELEMETRY
	S/C TO GND TERM	137.11	0.03	5	2	PRIME FOCUS	13°	17	14	RCP	ATS SATAN	15	20.3	+17.3	13.1	137.11	YES	BACK UP TELEMETRY
	S/C TO GND TERM	≈4000	40	20	28	HORN	20	16.7	13.7	LINEAR / LINEAR	85°P 40°P	0.21 0.46	56.0 49.5	+53.0 46.5	36.2 29.5	≈4000	YES	BACK UP TELEMETRY

RELEASE DATE: 10/15/68
REV. 1 10/24/68
REV. 2 11/15/68
REV. 3 12/6/68
REV. 4 3/15/69
REV. 5 10/6/69
REV. 6 11/5/69
REV. 7 2/24/71
REV. 8 6/4/71

PREPARED BY: [signature]
ATS F & G
GROUND TERMINAL SECTION
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*NOTE: MINIMUM SPACECRAFT ANTENNA GAINS OVER FIELD OF VIEW NOT DEFINED

APPENDIX B

ACRONYMS

ABC	ANALOG BACKUP CONTROLLER
ACE	ACTUATOR CONTROL ELECTRONICS
ACS	ATTITUDE CONTROL SUBSYSTEM
AFC	AUTOMATIC FREQUENCY CONTROL
AGC	AUTOMATIC GAIN CONTROL
AGE	AEROSPACE GROUND EQUIPMENT
AGREE	ADVISORY GROUP ON RELIABILITY ELECTRONIC EQUIPMENT
AM	AMPLITUDE MODULATED
AMP	AMPLIFIER
APS	AUXILIARY PROPULSION SUBSYSTEM
ASC	AEROSPACE CORPORATION
ASSY	ASSEMBLY
AST	ATLANTIC STANDARD TIME
ATC	AIR TRAFFIC CONTROL
ATS	APPLICATIONS TECHNOLOGY SATELLITE
ATS-R	APPLICATIONS TECHNOLOGY SATELLITE-RANGING
ATSOCC	ATS OPERATIONS CONTROL CENTER
BB	BASEBAND
BECO	BOOSTER ENGINE CUTOFF
BTE	BENCH TEST EQUIPMENT
BW	BANDWIDTH
CCIR	INTERNATIONAL RADIO CONSULTATIVE COMMITEE
CCITT	INTERNATIONAL TELEGRAPH AND TELEPHONE CONSULTATIVE COMMITTEE
CDE	COMMAND/DECODER DISTRIBUTOR
CEI	CONTRACT END ITEM
CETEC	COMMUNICATION EXPERIMENT TEST AND EVALUATION CONSOLE
CFE	CONTRACTOR FURNISHED EQUIPMENT
CMD	COMMAND
COMPUT	COMPUTING CENTER GSFC
COMSAT	COMMUNICATION SATELLITE
CR	CONTROL ROOM
CRB	CHANGE REVIEW BOARD
CSS	COARSE SUN SENSOR
CW	CARRIER WAVE
DACU	DATA ACQUISITION AND CONTROL UNIT
DAF	DATA ACQUISITION FACILITY
DHE	DATA HANDLING EQUIPMENT
DISC	DISCRIMINATOR
DLO	DOUBLE LOCAL OSCILLATOR
DOC	DIGITAL OPERATIONAL CONTROLLER
DOD	DEPARTMENT OF DEFENSE
DRFA	DATA REDUCTION AND ANALYSIS
DRSS	DATA RELAY SATELLITE SYSTEM
DSS	DIGITAL SUN SENSOR
DSU	DATA SWITCHING UNIT
DTL	DIODE-TRANSISTOR LOGIC
ECH	EARTH COVERAGE HORN
EDT	EASTERN DAYLIGHT TIME
EIRP	EFFECTIVE ISOTROPIC RADIATED POWER
EMC	ELECTROMAGNETIC COMPATIBILITY
EME	ENVIRONMENTAL MEASUREMENTS EXPERIMENT
EMI	ELECTROMAGNETIC INTERFERENCE
ERL	ENVIRONMENTAL RESEARCH LABORATORIES
ERP	EFFECTIVE RADIATED POWER

ES	EARTH SENSOR
ESSA	ENVIRONMENTAL SCIENCE SERVICES ADMINISTRATION
EST	EASTERN STANDARD TIME
ETP	ENGINEERING TEST PROCEDURE
ETR	EASTERN TEST RANGE
ETR	EASTERN TEST RANGE
EVM	EARTH VIEWING MODULE
FAA	FEDERAL AVIATION ADMINISTRATION
FDM	FREQUENCY DIVISION MULTIPLEX
FHC	FAIRCHILD-HILLER CORPORATION
FM	FREQUENCY MODULATED
FMECA	FAILURE MODE EFFECTS AND CRITICALITY ANALYSIS
FOV	FIELD OF VIEW
FSK	FREQUENCY SHIFT KEYING
FSS	FINE SUN SENSOR
FT	FREQUENCY TRANSLATION
GAC	GROUND ATTITUDE CONTROL
GEOS	GEODETIC EARTH ORBITING SATELLITE
GFE	GOVERNMENT FURNISHED EQUIPMENT
GFRP	GRAPHITE FIBER REINFORCED PLASTIC
GIE	GROUND INSTRUMENTATION EQUIPMENT
GMT	GREENWICH MEAN TIME
GND	GROUND
GSFC	GODDARD SPACE FLIGHT CENTER
HAC	HUGHES AIRCRAFT COMPANY
HR	HIGH RESOLUTION
ICD	INTERFACE CONTROL DOCUMENTATION FHC
IDCS	IMAGE DISSECTOR CAMERA SYSTEM
IF	INTERMEDIATE FREQUENCY
IGS	INERTIAL GUIDANCE SYSTEM
IMU	INERTIAL MEASUREMENT UNIT
INT	INTERFEROMETER
IPD	INFORMATION PROCESSING DIVISION
IRA	INERTIAL REFERENCE ASSEMBLY
I/O	INPUT/OUTPUT
ITL	INTEGRATE-TRANSFER-LAUNCH
ITU	INTERNATIONAL TELECOMMUNICATIONS UNION
ITV	INSTRUCTIONAL TELEVISION
LASER	LIGHT AMPLIFICATION BY STIMULATED EMISSION OF RADIATION
LASS	LATERAL ACCELERATION SENSING SYSTEM
LAST	LOW ALTITUDE SATELLITE POINTING
LC	LAUNCH COMPLEX
LCP	LEFT HAND CIRCULAR POLARIZED
LIC	LOAD INTERFACE CIRCUIT
LO	LOCAL OSCILLATOR
LOC	LAUNCH OPERATIONS CENTER
LOS	LINE OF SIGHT
L/V	LAUNCH VEHICLE
MDAC	MCDONNELL-DOUGLAS AIRCRAFT CORPORATION
MGC	MISSILE GUIDANCE COMPUTER
MIL	MILITARY SPEC OR STD
MMW	MILLIMETER WAVE EXPERIMENT
MOD	MODULATOR
MST	MOUNTAIN STANDARD TIME
MUX	MULTIPLEX

ATS Acronyms (Cont)

NASA	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
NASCOM	NASA COMMUNICATIONS NETWORK
NASCOP	NASA COMMUNICATIONS OPERATING PROCEDURES
NI-CD	NICKEL-CADMIUM
NM	NAUTICAL MILE
NOAA	NATIONAL OCEANIC AND ATMOSPHERIC ADMINISTRATION
NPR	NOISE POWER RATIO
NRZ	NON RETURN TO ZERO
OSI	OPERATIONS AND INSTRUMENTATION
OCC	OPERATION CONTROL COMPLEX
OMNI	OMNIDIRECTIONAL
OCCEN	OPERATION CONTROL CENTER FOR LAUNCH
OSC	OSCILLATOR
OSR	OPTICAL SOLAR REFLECTOR
PAM	PHASE AMPLITUDE MODULATION
PCE	PROPULSION CONTROL ELECTRONICS
PCM	PULSE CODE MODULATION
PCMDHE	PCM TELEMETRY DATA HANDLING EQUIPMENT
PCU	POWER CONTROL UNIT
PDT	PACIFIC DAYLIGHT TIME
PEP	PEAK ENVELOPE POWER
PFF	PRIME FOCUS FEED
PFM	PULSE FREQUENCY MODULATION
PLACE	POSITION LOCATION AND AIRCRAFT COMMUNICATIONS EXPERIMENT
PLF	PAYLOAD FAIRING
PM	PHASE MODULATED
PMS	PARABOLIC REFLECTOR SUBSYSTEM
PRU	POWER REGULATION UNIT
PS	POLARIS SENSOR
PST	PACIFIC STANDARD TIME
RELAP	RELIABILITY AND QUALITY ASSURANCE PROCEDURES
R&R	RANGE AND RANGE RATE
RCP	RIGHT HAND CIRCULAR POLARIZED
RCV	RECEIVE
RCVR	RECEIVER
REM	ROCKET ENGINE MODULES
RF	RADIO FREQUENCY
RFI	RADIO FREQUENCY INTERFERENCE
RFSW	RF SWITCH
RMU	REMOTE MULTIPLEX UNIT
SAGS	SMALL APERTURE GROUND STATION
SAMOC	SPACECRAFT ATTITUDE MANEUVERING OPTIMAL CONTROL
SAPSAC	SELF-ADAPTIVE PRECISION POINTING SPACECRAFT ATTITUDE CONTROL
SAS	SOLAR ASPECT SENSOR
SATAN	SATELLITE AUTOMATIC TRACKING ANTENNA NETWORK
SC	SIGNAL CONDITIONER
SC	SPACECRAFT CLOCK
SCAMA	SWITCHING, CONFERENCING AND MONITORING ARRANGEMENT
SCAMP	SMALL COMMAND ANTENNA MEDIUM POWER
SE	SYSTEM DESIGN EHC
SDL	SPECIAL DATA LINK
SEC	SYNCHRONOUS EQUATORIAL CIRCULAR ORBIT
SECO	SUSTAINER ENGINE CUTOFF
SHF	SUPER HIGH FREQUENCY
SLV	STANDARD LAUNCH VEHICLE

SMAB	SOLID MOTOR ASSEMBLY BUILDING
SMR	SWITCHING MODE REGULATOR
SNR	SIGNAL-TO-NOISE RATIO
SPS	SPACECRAFT PROPULSION SYSTEM
SRM	SOLID ROCKET MOTOR
S/C	SPACECRAFT
SSB	SINGLE SIDEBAND
STADAN	SPACE TRACKING AND DATA ACQUISITION NETWORK
SYNCOM	SYNCHRONOUS COMMUNICATION
T&C	TELEMETRY AND COMMAND
TDRSS	TRACKING AND DATA RELAY SATELLITE SYSTEM
TDS	TRACKING AND DATA SYSTEM DIRECTORATE
TEED	TIME AND FREQUENCY DISPERSION
TACO	TEXAS ANTENNA COMPANY
TBD	TO BE DECIDED
TCG	TIME CODE GENERATOR
TCU	TEMPERATURE CONTROL UNIT
TDA	TUNNEL DIODE AMPLIFIER
TDC	TRANSPONDER COMMAND DISTRIBUTION
TDM	TIME DIVISION MULTIPLEX
TGS	TRANSPORTABLE GROUND STATION
TIROS	TELEVISION AND INFRA RED OBSERVATION SATELLITE
TM	TELEMETRY
TRUST	TELEVISION RELAY USING SMALL TERMINALS
TT&C	TELEMETRY, TRACKING, AND COMMAND
TTY	TELETYPE
TV	TELEVISION
TVC	THRUST VECTOR CONTROL
TWT	TRAVELING WAVE TUBE
TWTA	TRAVELING WAVE TUBE AMPLIFIER
TWX	TELETYPE
UCLA	UNIVERSITY OF CALIFORNIA AT LOS ANGELES
UCSD	UNIVERSITY OF CALIFORNIA AT SAN DIEGO
UDMH	UNSYMMETRICAL DIMETHYLHYDRAZINE
UES	UNIVERSAL ENVIRONMENTAL SHELTER
UHF	ULTRA HIGH FREQUENCY
UNH	UNIVERSITY OF NEW HAMPSHIRE
UT	UNIVERSAL TIME
VCO	VOLTAGE CONTROLLED OSCILLATOR
VCXO	VOLTAGE CONTROLLED CRYSTAL OSCILLATOR
VECO	VERNIER ENGINE CUTOFF
VHF	VERY HIGH FREQUENCY
VHR	VERY HIGH RESOLUTION
VIB	VERTICAL INTEGRATION BUILDING
VLF	VERY LOW FREQUENCY
VSWR	VOLTAGE STANDING WAVE RATIO
XMTR	TRANSMITTER

SMAB	SOLID MOTOR ASSEMBLY BUILDING
SMR	SWITCHING MODE REGULATOR
SNR	SIGNAL-TO-NOISE RATIO
SPS	SPACECRAFT PROPULSION SYSTEM
SRM	SOLID ROCKET MOTOR
S/C	SPACECRAFT
SSB	SINGLE SIDEBAND
STADAN	SPACE TRACKING AND DATA ACQUISITION NETWORK
SYNCOM	SYNCHRONOUS COMMUNICATION
TEC	TELEMETRY AND COMMAND
TDRSS	TRACKING AND DATA RELAY SATELLITE SYSTEM
TDS	TRACKING AND DATA SYSTEM DIRECTORATE
TFD	TIME AND FREQUENCY DISPERSION
TACO	TEXAS ANTENNA COMPANY
TBD	TO BE DECIDED
TCG	TIME CODE GENERATOR
TCU	TEMPERATURE CONTROL UNIT
TDA	TUNNEL DIODE AMPLIFIER
TDC	TRANSPONDER COMMAND DISTRIBUTION
TDM	TIME DIVISION MULTIPLEX
TGS	TRANSPORTABLE GROUND STATION
TIROS	TELEVISION AND INFRA RED OBSERVATION SATELLITE
TM	TELEMETRY
TRUST	TELEVISION RELAY USING SMALL TERMINALS
TTCC	TELEMETRY, TRACKING, AND COMMAND
TTY	TELETYPE
TV	TELEVISION
TVC	THRUST VECTOR CONTROL
TWT	TRAVELING WAVE TUBE
TWTA	TRAVELING WAVE TUBE AMPLIFIER
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UCLA	UNIVERSITY OF CALIFORNIA AT LOS ANGELES
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UT	UNIVERSAL TIME
VCO	VOLTAGE CONTROLLED OSCILLATOR
VCXO	VOLTAGE CONTROLLED CRYSTAL OSCILLATOR
VECO	VERNIER ENGINE CUTOFF
VHF	VERY HIGH FREQUENCY
VHR	VERY HIGH RESOLUTION
VIB	VERTICAL INTEGRATION BUILDING
VLF	VERY LOW FREQUENCY
VSWR	VOLTAGE STANDING WAVE RATIO
XMTR	TRANSMITTER